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LST PHASE A DESIGN UPDATE STUDY

By Program Development

August 1, 1973

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16. ABSTRACT

This document is an update of the Phase A study (TM X-64726) of the Large Space Telescope (LST), based on changes in guidelines and new data developed subsequent to the Phase A study. The study defines an LST concept based on the broad mission guidelines provided by the Office of Space Science (OSS), the scientific requirements developed by OSS with the scientific community, and an understanding of long range NASA planning current at the time the study was performed.

The LST is an unmanned astronomical observatory facility, consisting of an optical telescope assembly (OTA), scientific instruments (SI), and a support systems module (SSM). The report consists of one volume. The report describes the constraints and trade off analyses that were performed to arrive at a reference design for each system and for the overall LST.

The LST will be launched into low earth orbit by the Space Shuttle and operated for 10 to 15 years. The Shuttle will also be used to maintain the LST and to update the scientific instrument complement. Ground-return maintenance is the primary maintenance mode, with limited on-orbit EVA as an adjunct.

The LST will provide the scientific community with several fundamentally unique capabilities which will permit the acquisition of new and important observational data. Its location in space permits observations over the entire spectrum from about 100 nm to the far infrared.

A low cost design approach was followed in the study. This resulted in the use of standard spacecraft hardware, the provision for maintenance at the black box level, growth potential in systems designs, and the sharing of Shuttle maintenance flights with other payloads.

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Please refer to NASA TM X-64726, Large Space Telescope Phase A Final Report, dated December 15, 1972.

TECHNICAL MEMORANDUM X-64763

LST PHASE A DESIGN UPDATE STUDY

CHAPTER I. INTRODUCTION

A. Objective

The objective of the study was to provide an updated LST design based on (1) changed guidelines, and (2) data developed within and outside of the Phase A study which became known too late to modify that study. It was desired to have the study results in time to give them to the OTA/SI Phase B study contractors as they began their studies.

B. Approach

To determine the nature of the preferred configuration and the degree of limited on-orbit maintenance to be allowed (see Chapter II), several concepts were compared, ranging from pure earth return maintenance concepts to extensive on-orbit maintenance concepts. To compare maintenance approaches and configuration concepts most accurately, a pure earth return maintenance concept was used as a comparison baseline, and the degree of impact caused by going to each of the other concepts was assessed in each discipline area. The primary emphasis in this assessment was on the concepts involving minor on-orbit maintenance, with a lesser effort being placed on the concepts which utilize more extensive on-orbit maintenance.

Broad definition of the levels of maintenance considered in assessing the configurations are included as Table I-1. The relative priorities of instruments are defined in the OTA/SI work statement guidelines. The complement of instruments and their arrangement as defined in the Phase A study were utilized. However, since this area is to be studied in the Phase B time frame also, the SSM configuration was not allowed to be driven to an extreme case based on this instrument arrangement. Rather, if some minor changes in instrument arrangement (such as moving the radial instruments closer to the external wall of the SSM to allow radial extraction through a hatch without requiring a long reach) would provide significant configuration benefit, these changes were considered permissible.

TABLE I-1. LEVELS OF MAINTENANCE

• Minor[a] - Replacement of failed and life-limited systems equipment and instrument sensors.
• Moderate - Replacement of failed, life-limited, and obsolete systems equipment and instrument sensors, and some entire instruments (probably approximately each 2 1/2 yr).
• Major Overhaul - Replacement of failed, life-limited, and obsolete systems equipment and instrument sensors, entire instruments, resurfacing of thermal control surfaces, repair of micrometeoroid punctures, recoating of optics, replacement of equipment which is not easily replaced on orbit (for example, solar arrays, secondary mirror mechanisms, etc.), recleaning and more extensive testing, etc. (probably each 5 yr).

a. Emergency maintenance is a less extensive version of minor maintenance, which also permits operation under more relaxed constraints than routine maintenance should permit (relaxed contamination constraints, for example).

C. Summary

An updated Phase A design was selected from several alternative approaches. This design provides a ground return maintenance approach as the primary maintenance mode, but allows up to a fairly high degree of on-orbit EVA maintenance with minimum impact on the LST design. The key design changes are summarized herein, with reasons why they were made. The changes in power requirements, mass characteristics, and systems performance are provided for the systems which were changed.

Timelines for the maintenance operations are provided, and estimates of the required spares and logistics are given. The estimated changes in reliability are given, and some of the key contamination control experience from the Apollo/Skylab program which is applicable to LST is provided.

The recommendations for the Phase B study which were listed in the Phase A Report are updated, where applicable, as a result of this study.

CHAPTER II. GUIDELINES

The guidelines provided in the OTA/SI Phase B study work statement were utilized for this study. The key changes in guidelines from the Phase A study are provided in Table II-1. Some of the changes in guidelines may require changes in concept of the OTA and/or SI. In such cases, they will be treated extensively during the Phase B study and were not dealt with in this study.

This study deals primarily with the SSM changes, although changes to the OTA and SI are suggested in a few areas.

TABLE II-1. KEY GUIDELINE CHANGES FROM PHASE A STUDY

1. Commonality with HEAO hardware is still desirable, but must be reevaluated, since the HEAO program has undergone rather significant changes recently. Potential commonality with other programs must be investigated and should be utilized where cost effective.
2. LST "lifetime" is 15 yr.
3. There is only one flight article. It must be refurbished, modified, etc., to achieve a 15-yr program life.
4. Earth-return maintenance is the primary mode to be used on the initial flight (2 1/2 yr lifetime prior to first return). The study will determine the impact of performing very limited on-orbit maintenance in addition to the earth-return mode. The study will also determine the impact of utilizing on-orbit maintenance more extensively after the initial earth return.
5. Low cost approaches are even more paramount than before, and must be utilized in all areas. Effects on initial LST cost (DDT and E) and operational costs must be considered separately; the effect on the latter of minimizing the former shall be identifiable, and vice versa.
6. There is no longer the requirement to fly on the Titan as a back-up to the Shuttle launch. However, the approximate dimensions and weight constraints imposed by the Viking shroud should be used as a general guide to bound the LST, unless it can be shown to be more cost effective to deviate further from these. This approach will help minimize impacts on the LST caused by any unforeseen reduction in Shuttle capabilities.
7. Avoidance cone angles for normal telescope viewing are now 5 deg from the earth's limb, 10 deg from the moon's center, and TBD for the sun as a function of the light shield truncation angle.
8. LST off-sun roll capability for spectrograph slit orientation shall be be maximum of once per day for a duration of less than 3 orbits; this shall not dominate the design.
9. The origin for the reference axes for mass locations, etc., is at a point on the optical axis where the axis intersects the plane containing the aft surface of the primary mirror ring.[a]
10. As a design goal, the LST should be able to view objects such as planets and comets.

a. This guideline must be changed in the Phase B study, since the ring has been replaced with a different structure in the Phase A update design.

CHAPTER III. MISSION ANALYSIS AND OPERATIONS

A. Shuttle Performance

The performance capability of the Space Shuttle has been revised since publication of the Phase A document. This revised capability of the Shuttle is presented in Figure III-1 as a function of circular orbital altitude at various altitudes. This data is taken from Space Shuttle System Payload Accommodations. JSC 07700, Vol. XIV, April 13, 1973.

B. Ground Maintenance

In addition to the following, Chapter VI also has a discussion on ground maintenance.

The LST Ground Return Maintenance Operation begins with the Shuttle Orbiter boosting the LST to parking orbit and circularizing in the typical lift-off to insertion mode. Following the performance of any shared mission that may be scheduled for flight, the Orbiter transfers to the LST orbit and approaches the LST for rendezvous. The LST has previously been commanded to standby with its telescope aperture doors closed, light shield retracted, and solar panels retracted. The Orbiter does not dock with the LST, but approaches to near vicinity where the Remote Manipulator System (RMS) is used to grasp the LST and stow it aboard the Orbiter. The LST is then deactivated, and the Orbiter is returned and landed at KSC. A timeline for this operation is shown on Figure III-2.

Following ground inspection and preparation, the LST is loaded aboard the Guppy aircraft and flown to a central integration-to-maintenance facility for refurbishment.

The major refurbishment activities, which take place concurrently, include recoating the optics, recoating the thermal control surfaces, repairing meteoroid damage to the meteoroid shield, major recleaning, replacing failed or life-limited items and changing out detectors and updating major instruments. These activities take approximately 2 weeks. The components are then inspected, shipped back to the integration and maintenance site and assembled under clean room conditions. Following inspection and verification of system integrity and cleanliness, the LST is flown by Guppy back to KSC. A timeline for this operation is shown on Figure III-2, also.

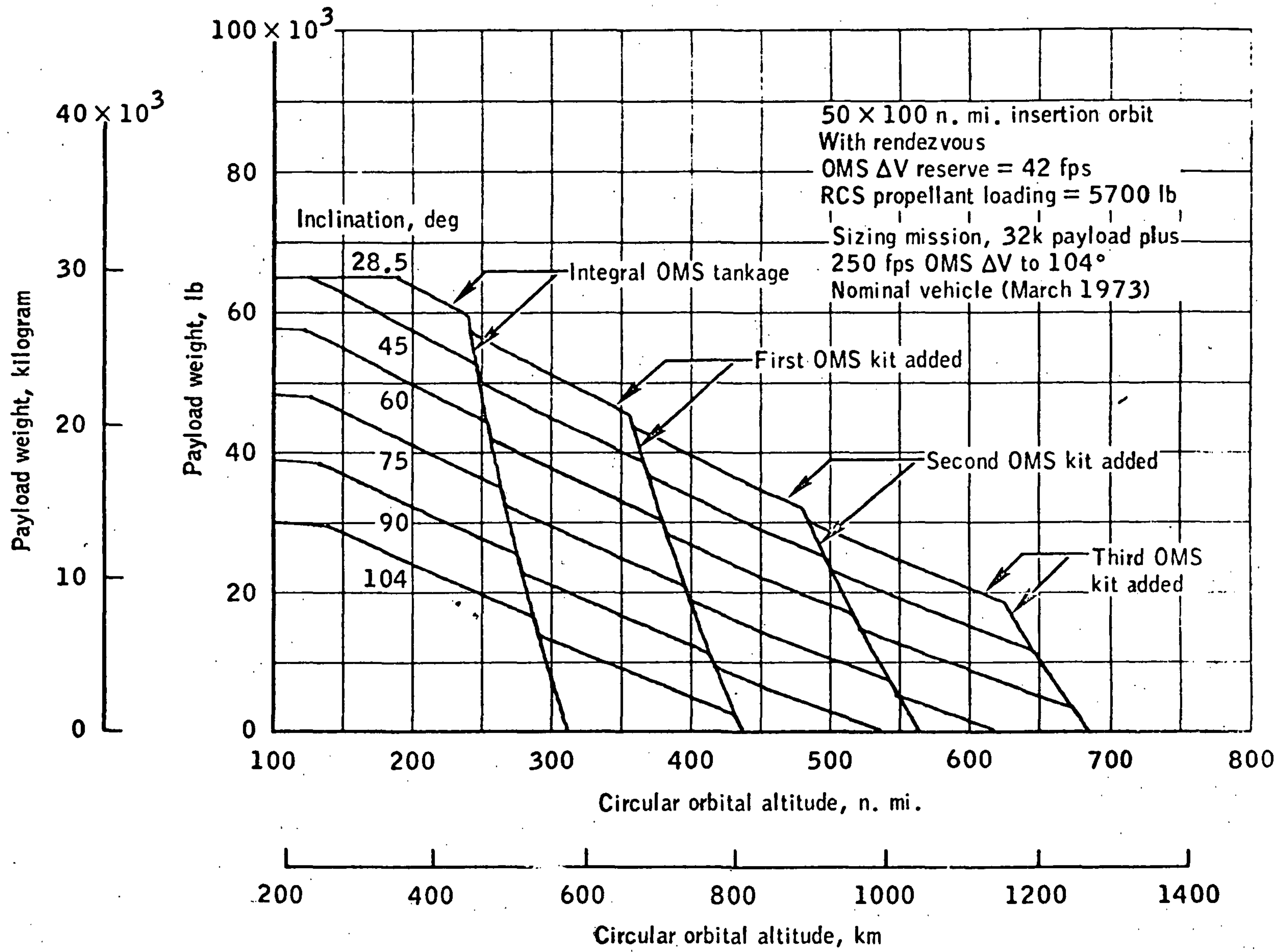


Figure III-1. Payload weight versus circular orbital altitude at various inclinations--delivery and rendezvous

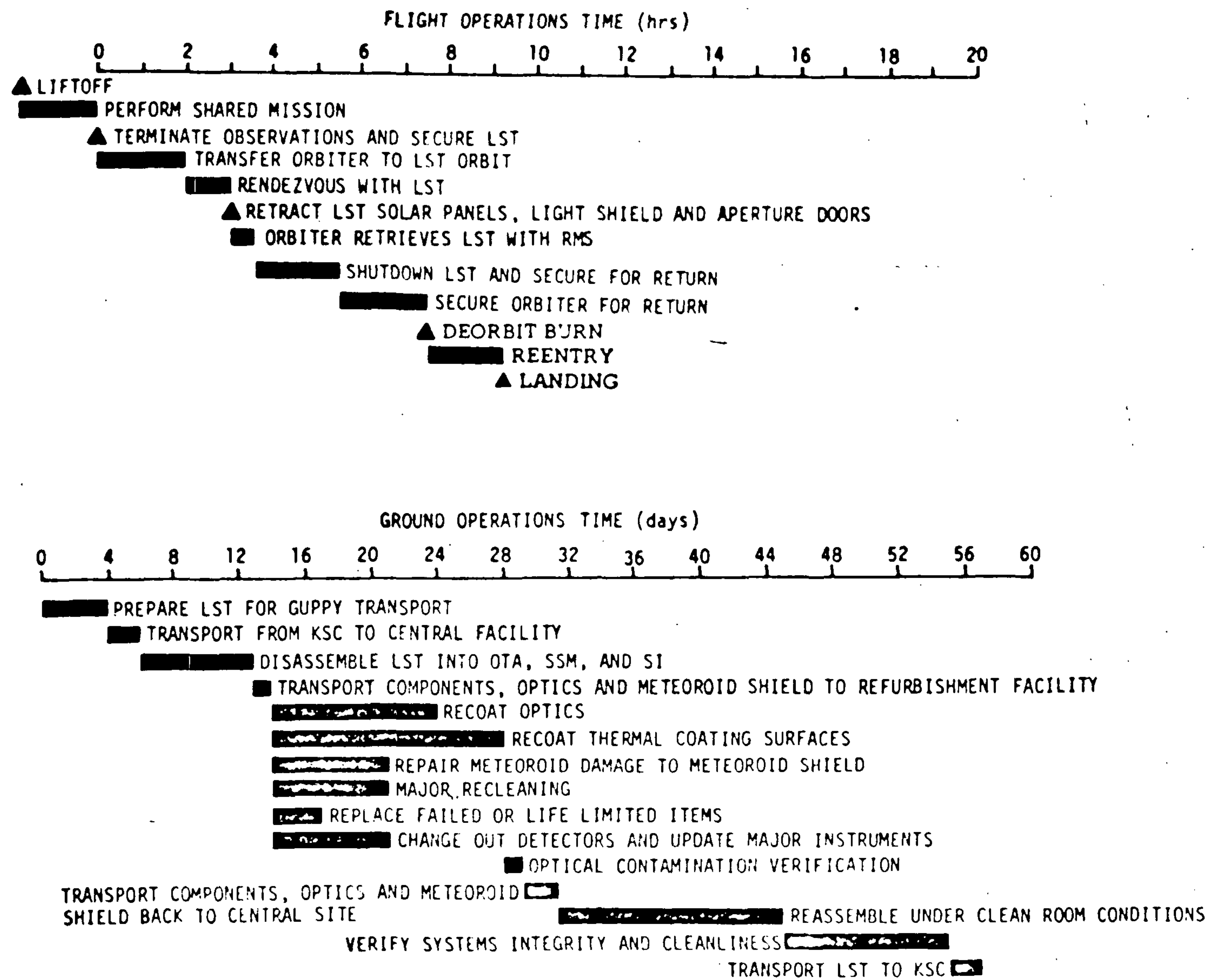


Figure III-2. Ground return maintenance option.

C. Minor On-Orbit EVA Maintenance

Minor on-orbit maintenance was determined in the study to be feasible, although ground return maintenance is the primary mode. A discussion on this maintenance is provided in Chapter VI. The list of spares provided in Chapter VI is more nearly a shopping list of potential replacement items than a typical maintenance load. The entire list was used, however, in developing the timeline for minor on-orbit EVA maintenance, to provide a worst case situation.

Minor on-orbit maintenance will require about 38 hours of on-orbit maintenance time and will consist of replacing the items indicated in the Service LST portion of Figure III-3. This mission will probably be performed as a shared mission with another payload which could itself consume up to about 34 hours mission time, assuming this LST timeline was utilized, without exceeding the 7-day nominal Shuttle mission time.

D. Minor On-Orbit RMS Maintenance

A discussion of on-orbit manipulator maintenance is provided in Chapter VI.

Figure III-4 shows a timeline for performing minor on-orbit maintenance utilizing the Shuttle-provided RMS. This approach requires the replacement of 17 trays, each of which contains various subsystem elements. The equipment or truss is the same as the SSM equipment listed in Figure III-3. (No telescope or science instruments are on trays.) Tray replacement will require approximately 12 hours (including crew rest) to complete, and must be followed by EVA for instrument change out which will require about 24 hours (including crew rest). Total time to perform actual service on LST is about 36 hours.

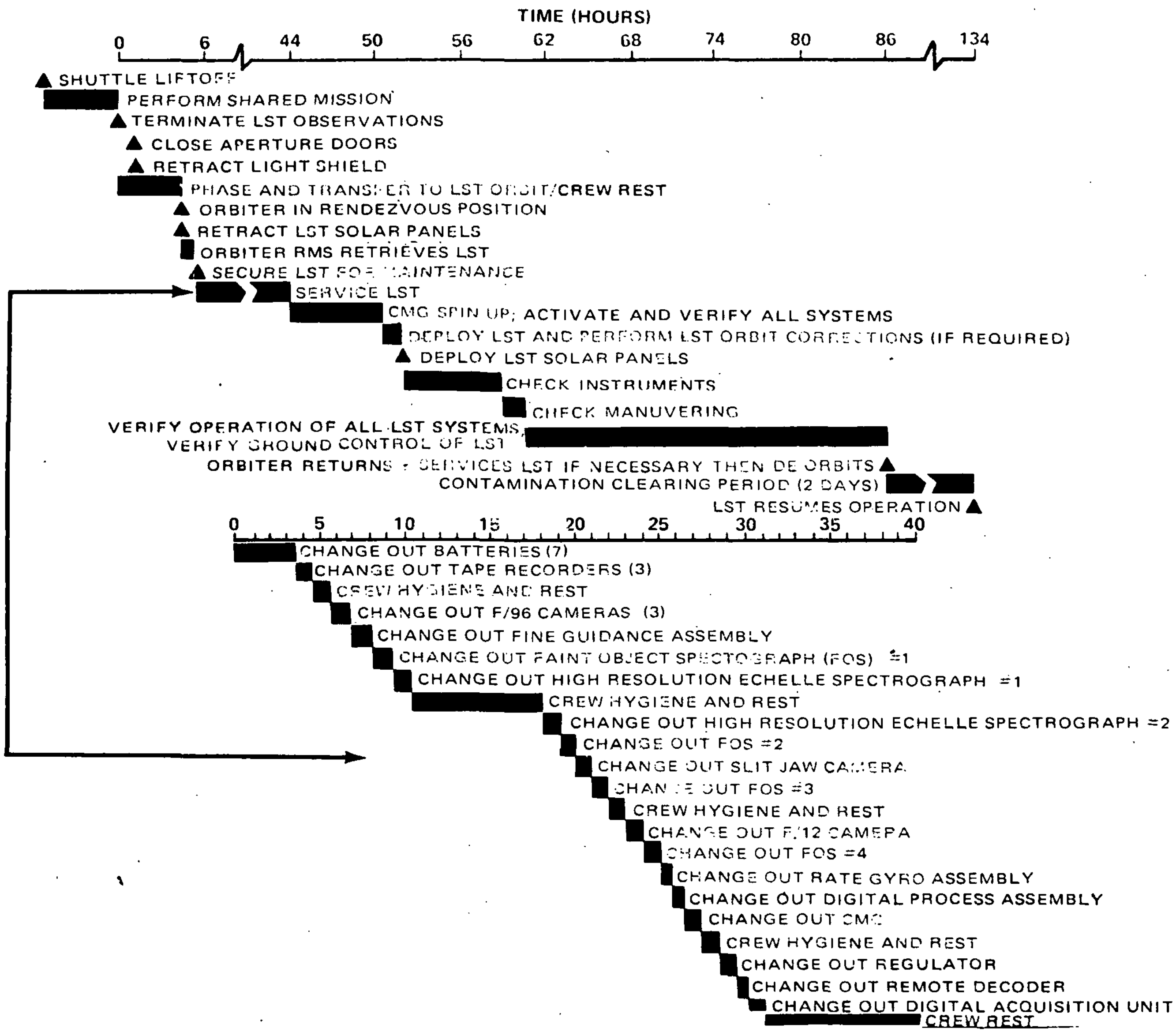


Figure III-3. Minor On-orbit maintenance EVA.

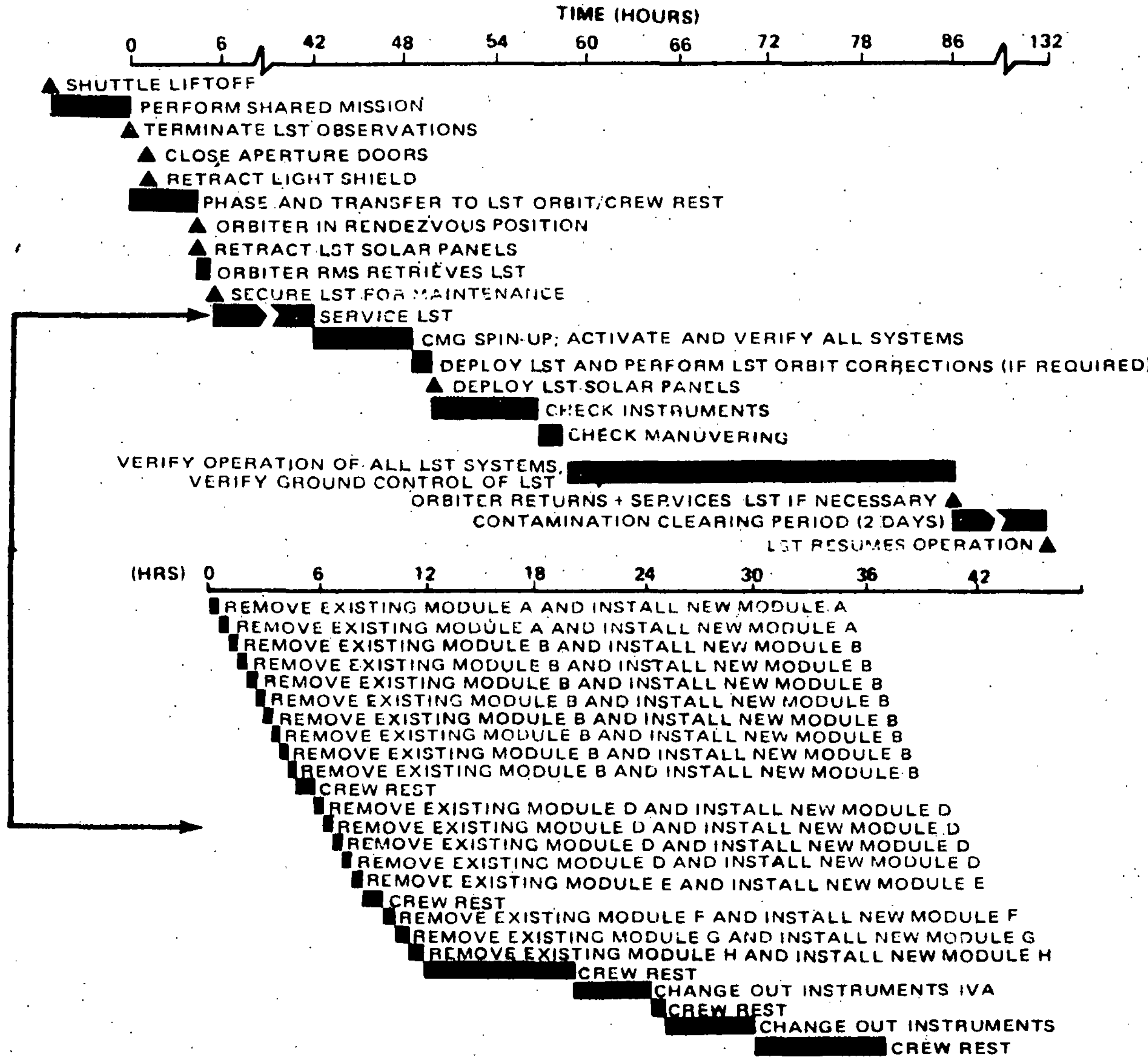


Figure III-4. Minor on-orbit maintenance with RMS.

CHAPTER IV. CONFIGURATION AND OVERALL SYSTEMS

A. Configuration Spectrum

Several key changes in guidelines have occurred since the Phase A study was completed. One of these was the addition of a maintenance mode guideline which dictated earth-return maintenance as the primary mode, with limited pressure-suited on-orbit maintenance as a potential adjunct to this mode. It was also desired to further explore the concept of utilizing the Shuttle manipulator for any limited on-orbit maintenance; hence this was also considered. Early in the study it was determined that the design did not appear to be constrained significantly by earth-return maintenance considerations, and hence a good portion of the analysis was directed to evaluating the limited on-orbit maintenance design impacts. Chapter VI provides a discussion on the maintenance-related design impacts.

To compare the concepts most effectively, several pure earth-return maintenance configurations were developed, and the impacts of adding limited on-orbit maintenance capability to these configurations were compared. Also, considerations of later adding more extensive on-orbit maintenance capability to these configurations were assessed. A broad spectrum of configurations, which was considered representative of the full range of possibilities for maintenance concepts, is shown in Figure IV-1. This study concentrated mainly on the concepts shown in the columns denoted pure earth-return maintenance and minor on-orbit maintenance, with the others serving as guides for assessing growth potential in degree of on-orbit maintenance. The configuration variables associated with these concepts are listed in Table IV-1. Figure IV-2 categorizes these concepts by basic structure categories.

For comparison purposes, the configurations were grouped into three basic types (truss, shell, and unitized SSM) within the maintenance modes, and comparisons were made on this basis. The truss, unitized SSM, and shell concepts are described in Sections C, D, and E, respectively.

B. Tray Mounting Scheme

A tray-mounted packaging scheme (Figs. IV-3 through -6) was developed for use by either a suited astronaut or the Shuttle manipulator. The tray (or some similar device) is required for manipulator maintenance, and also is advantageous for the suited astronaut if a large quantity of

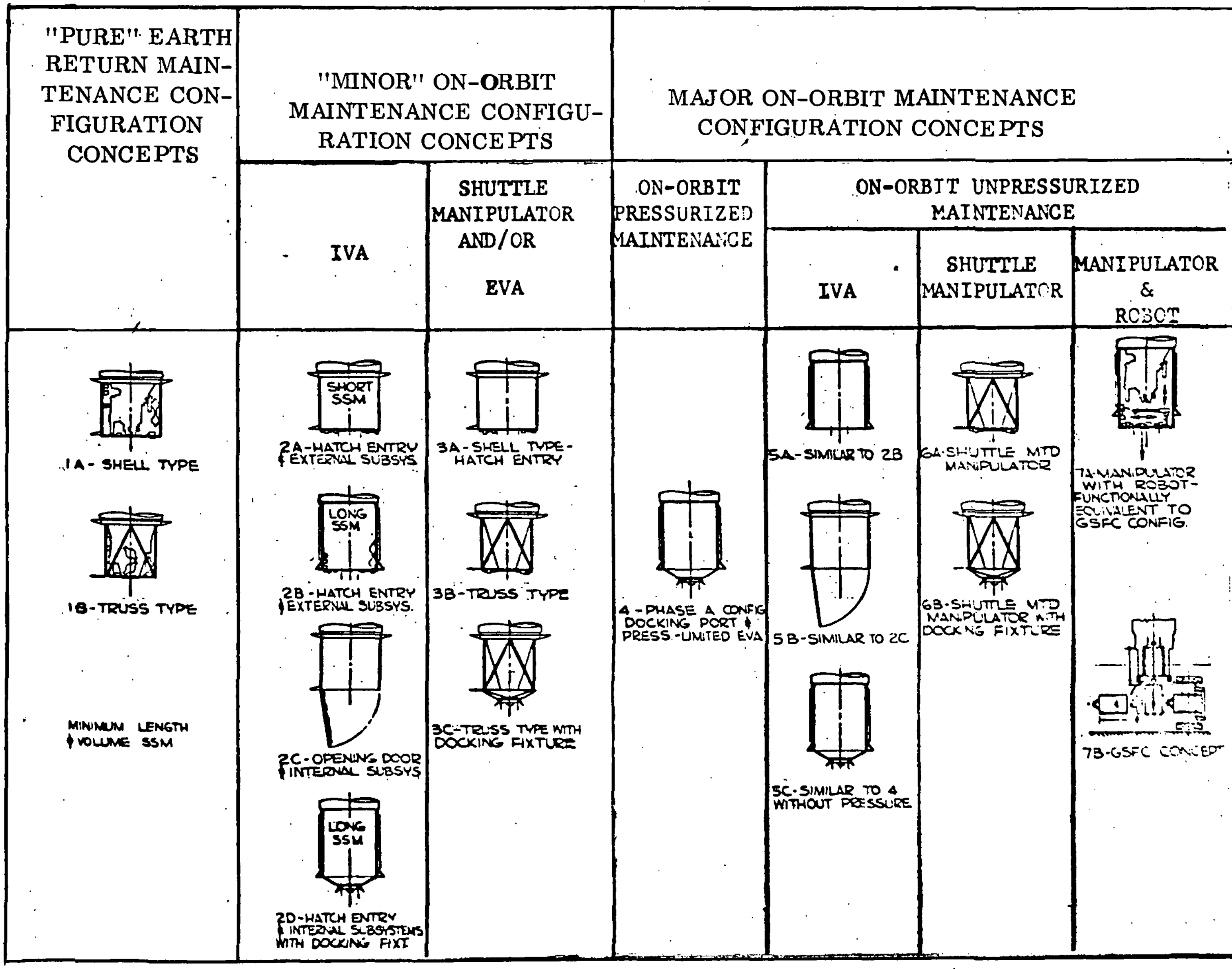


Figure IV-1. LST configuration concepts.

TABLE IV-1. LST CONFIGURATION VARIABLES

1. Structure
• Shell
• Truss
• Truss with nonload-bearing shell (no concept shown)
• Unitized SSM
2. Maintenance Mode
• Earth-Return
• On-Orbit
  • Pressurized
  • EVA
  • Manipulator
  • Internal robot
  • External robot
  • Hybrids
3. Degree of On-Orbit Maintenance
• None
• Minor
• Extensive
4. Modularity/Package Level
• Component
• Tray
• Saddle-bag or box (no concept shown)
5. Location of Components
• External
• Internal
6. Docking/Holding Technique[a]
• Latched into bay (launch locks)
• Docking module
7. Access Means
• Axial hatch
• Hinged rear plate
• Side hatches (no concept shown)
• Side access (no hatches)
8. Solar Array
• Rigid panels
• Rollup array

a. Holding the LST with only the Shuttle manipulator during maintenance operations was considered too risky.

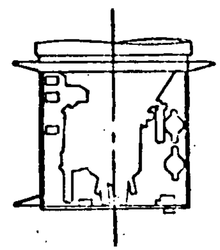
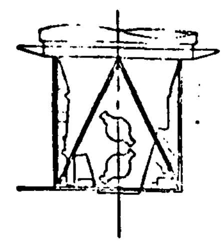
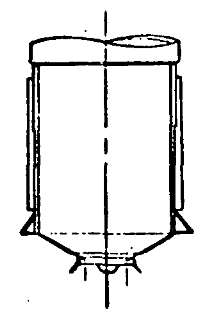
TYPE SSM / MAINTENANCE MODE	TYPE I - SHORT CYLINDRICAL SSM	TYPE II - TRUSS STRUCTURE SSM	TYPE III - LONG CYLINDRICAL SSM
MAINTENANCE CONFIGURATIONS - LISTED BY NUMBER	 1A 2A 3A	 1B 3B 6A	 4 2D 5C
DERIVATIVE CONFIGURATIONS	2C 5B	3C 6B	2B 5A 7A

Figure IV-2. SSM basic structural configurations - LST.
(GSFC configuration not shown)

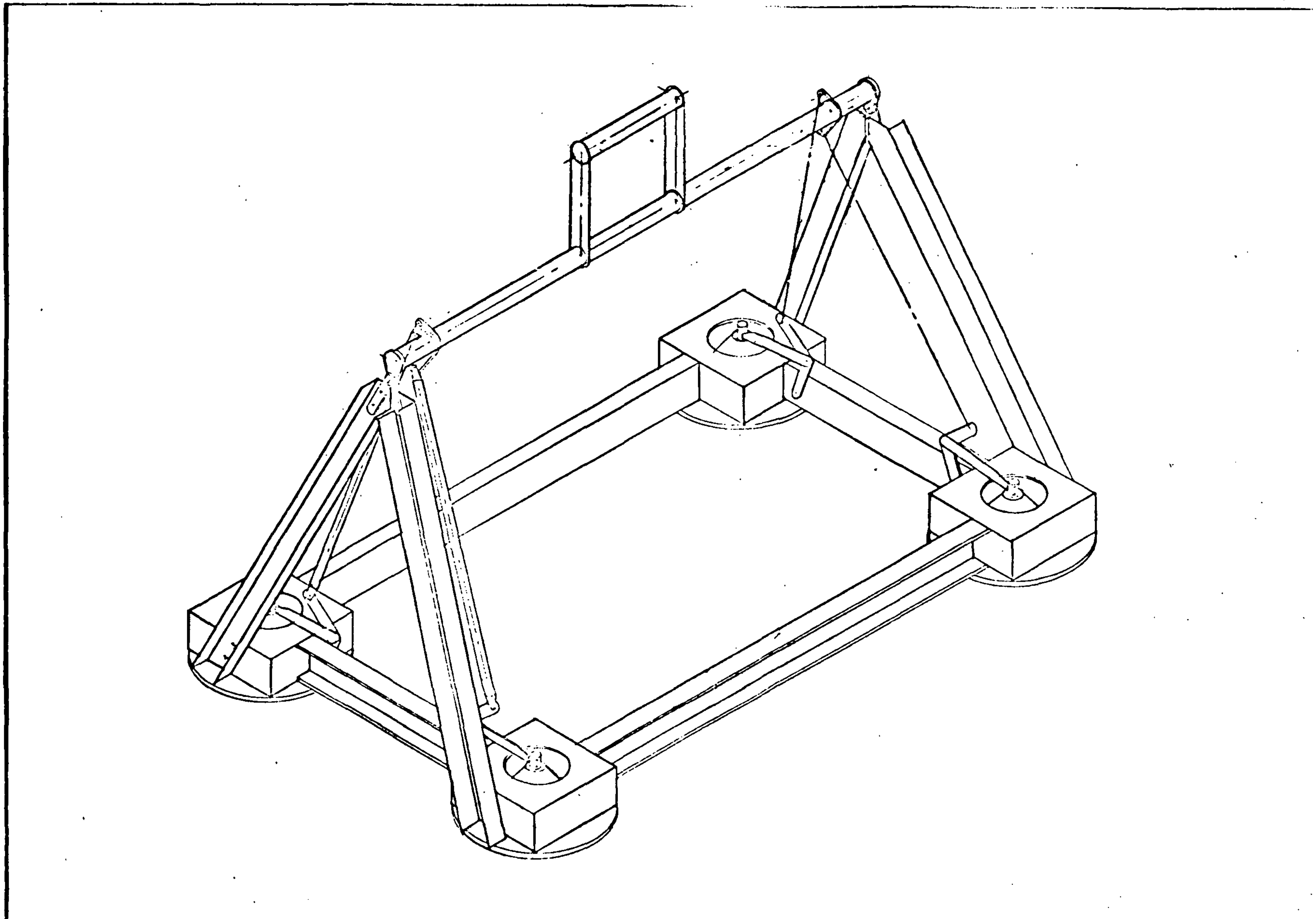


Figure IV-3. Subsystems support tray general view.

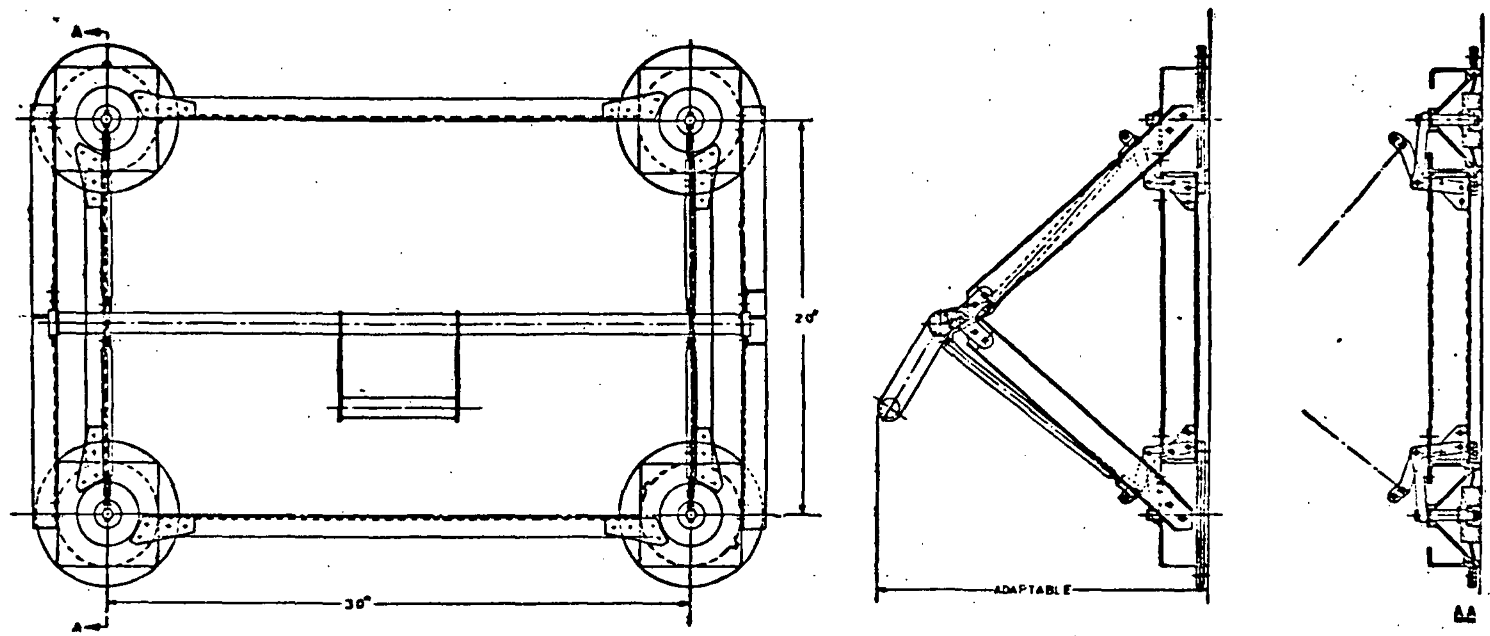


Figure IV-4. Subsystems support tray general arrangement.

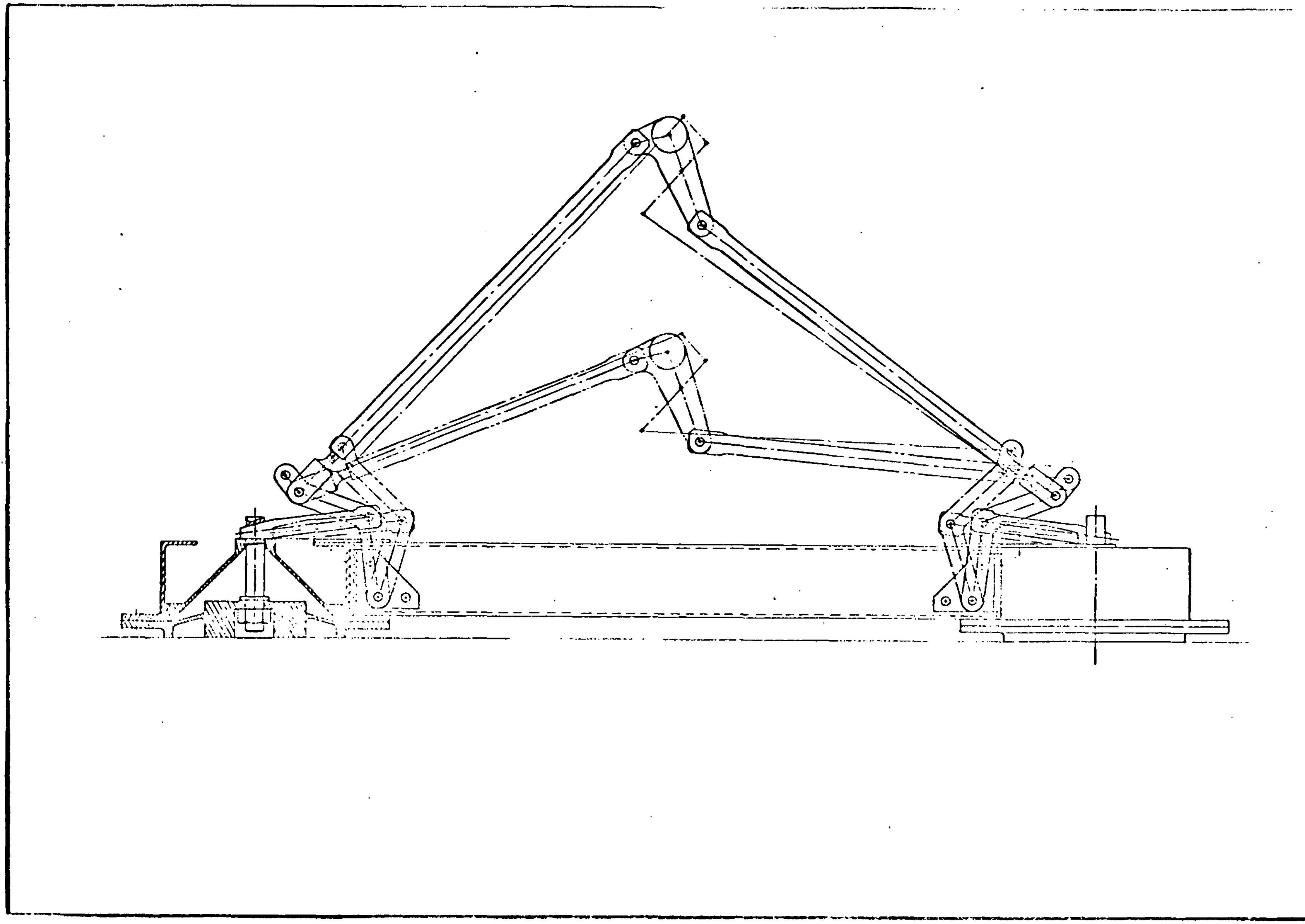


Figure IV-5. Subsystems support tray locking mechanism. (height variable)

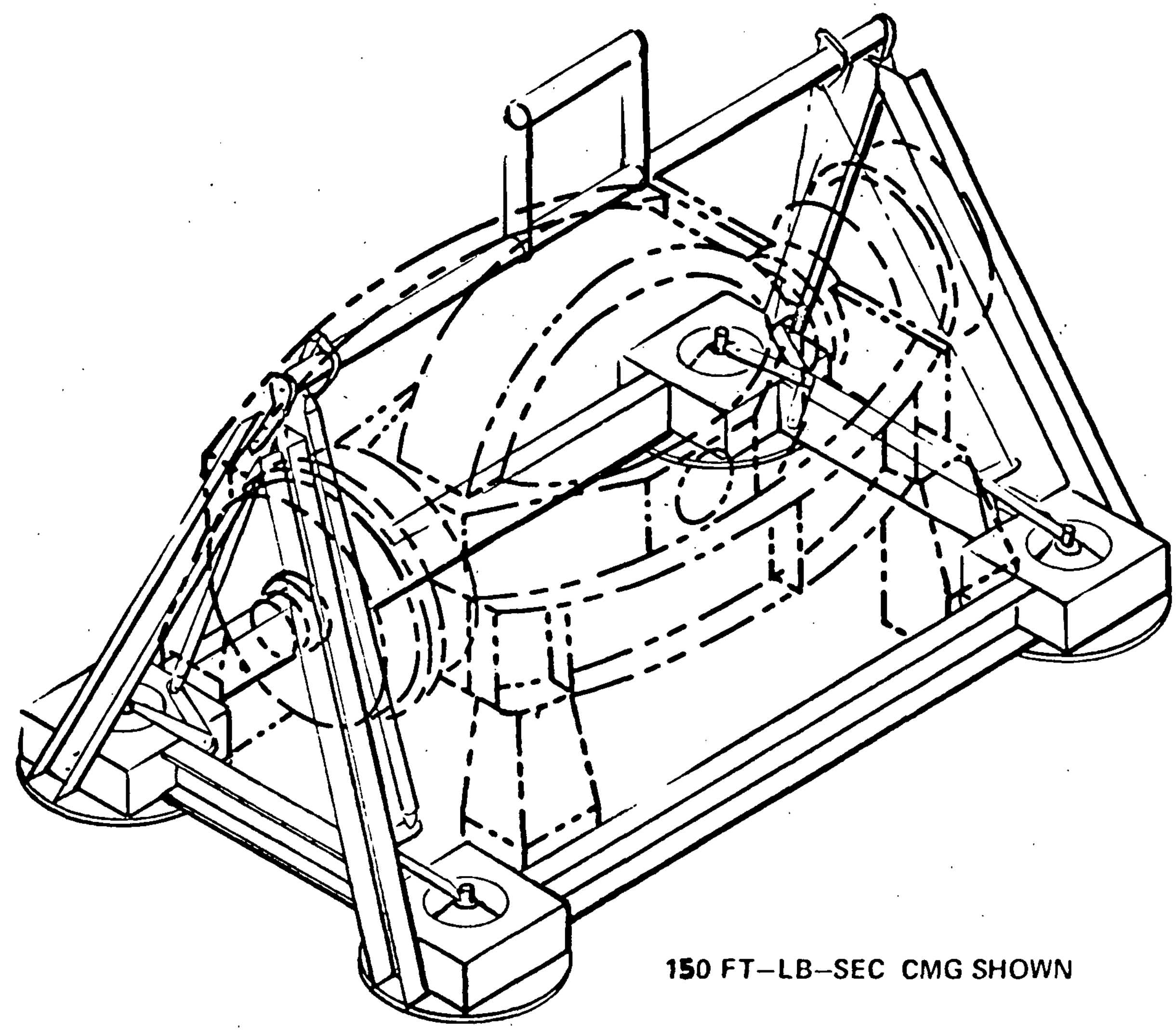


Figure IV-6. Subsystems support with CMG.

equipment must be replaced on-orbit. Basically the same tray design can be utilized for either manipulator or suited maintenance, and the decision on the actual maintenance mode to be used can be delayed much longer. The manipulator maintenance being considered would utilize the Shuttle manipulator with no modifications (if possible) to keep down costs of end-effectors, special alignment and handling devices, etc. The package sizes of the equipment must be kept small (keeping connector forces and latch/unlatch preload forces small), since the manipulator force capability is only about 4.5 kg (10 lb). The tray-mounted packages investigated during the study included the SSM equipment which was deemed to be most likely to fail, but did not include any SI equipment. The basic tray (Figs. IV-3 through IV-5) was designed to handle the largest module, the CMG, as shown by Figure IV-6. Nine different types of modules (Fig. IV-6A) were configured, omitting the RCS modules which were undefined at that time (Modules C and I). The rationale for modularization was to functionally group the systems to fit a minimum number of type of modules of four different sizes. From the Phase A report reliability analysis it was determined that the nine modules of Figure IV-6A would suffice. The RGA and FST were omitted from consideration for modular replacement because it was felt to be too difficult to replace these precisely and rigidly enough using the tray concept — EVA or ground-return was groundruled for these items. Figure IV-5 shows how the tray height may be varied to match the subsystems mounted on the trays.

C. Open Truss Configuration

The open truss offers maximum access to the manipulator. Several configurations were developed (Figs. IV-7 through IV-12). Nearly all the subsystems, including the CMGs, could be mounted on tray-type modules, and by judicious placement of the trays, all trays could be reached by the manipulator on three of the configurations (Figs. IV-9, 10, and 11). As Figure IV-12 shows, the addition of a docking assembly to the open truss results in very limited manipulator access, since at least two of the CMG modules must be mounted in an area inaccessible to the manipulator. Figures IV-8 through IV-11 illustrate the spherical volume inaccessible to the manipulator for the SSM-forward-mounted position of the LST in the Shuttle payload bay.

The open truss configuration utilizes the Primary Frame Assembly described in Section E as its primary structural reference member. The latest Shuttle attachment fitting spacing dimensions and arrangement were not available until too late to include on any but the recommended shell

MODULE A (2) REQ'D (POWER):	kg	lb	kg	lb
(1) 36 ELECTRICAL CONTROL ASSEMBLY	7.26	16		
(3) 31 REGULATORS	20.4	45		
TRAY	11.3	25		
TOTAL	39	86	78	172
MODULE B (6) REQ'D (POWER):				
(1) 29 BATTERY	22.4	49.3		
(1) 30 CHARGER	6.8	15		
TRAY	11.3	25		
TOTAL	40.5	89.3	243	535.8
MODULE C (1) REQ'D (RCS):				
(1) 8 LINES, VALVES, & REGULATOR SET	43.9	96.7		
(NOT PRESENTLY SHOWN)				
TRAY	11.3	25		
TOTAL	55.2	121.7	55.2	121.7
MODULE D (4) REQ'D (ACS):				
(1) 1 CMG	45.4	100		
TRAY	11.3	25		
TOTAL	56.7	125	226.8	500
MODULE E (1) REQ'D (ACS):				
(1) 6 *MAGNETIC TORQUER* ELECTRONICS	2.27	5		
(1) 11 DIGITAL PROCESSOR ASSY.	6.2	13.7		
(2) 12 TRANSFER ASSY	9.3	20.6		
TRAY	11.3	25		
TOTAL	29.2	64.3	29.2	64.3

MODULE F (1) REQ'D (C&DH):	kg	lb	kg	lb
(3) 26 TAPE RECORDERS	16.3	36		
TRAY	11.3	25		
TOTAL	27.6	61	27.6	61
MODULE G (1) REQ'D (C&DH):				
(1) 10 APOLLO TRANSPONDER	14.5	32		
(1) 17 ERTS TRANSPONDER	10.9	24		
TRAY	11.3	25		
TOTAL	36.7	81	36.7	81
MODULE H (1) REQ'D (C&DH):				
(1) 18 PSK DEMODULATOR	0.9	2		
(2) 19 DATA CONTROL UNIT	3.6	8		
(2) 20 FORMAT GENERATOR	3.6	8		
TRAY	11.3	25		
TOTAL	19.5	43	19.5	43
MODULE I (1) REQ'D (RCS):				
(1) 15 RCS TANK	32.3	71.3		
GN_2	19.5	43		
(NOT PRESENTLY SHOWN)				
TRAY	11.3	25		
TOTAL	63.2	139.3	63.2	139.3

TOTAL 779.3 kg (1718.1 lb), [660.9 kg (1457.1 lb) PRESENTLY SHOWN]

NOTES:
1. NUMBERS ABOVE IN PARENTHESES ARE THE QUANTITY. OTHER NUMBERS ARE IDENTIFICATION NUMBERS ON PHASE A MASTER EQUIPMENT LIST.
2. 9 TYPES OF MODULES, 7 PRESENTLY SHOWN
   18 TOTAL MODULES, 16 PRESENTLY SHOWN

Figure IV-6A. SSM subsystems modularization.

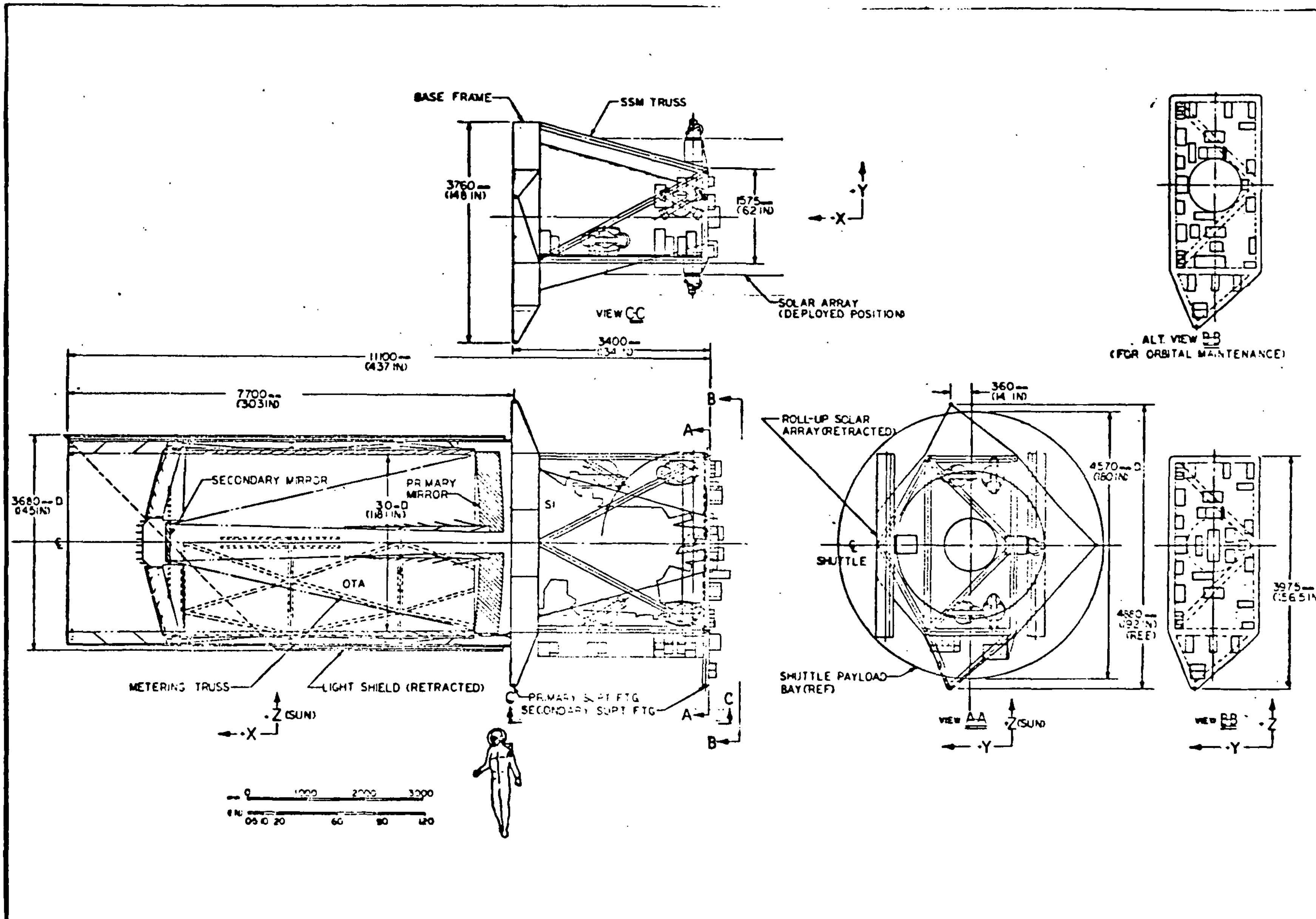


Figure IV-7. Configuration 1B - truss type SSM / LST earth return maintenance only.

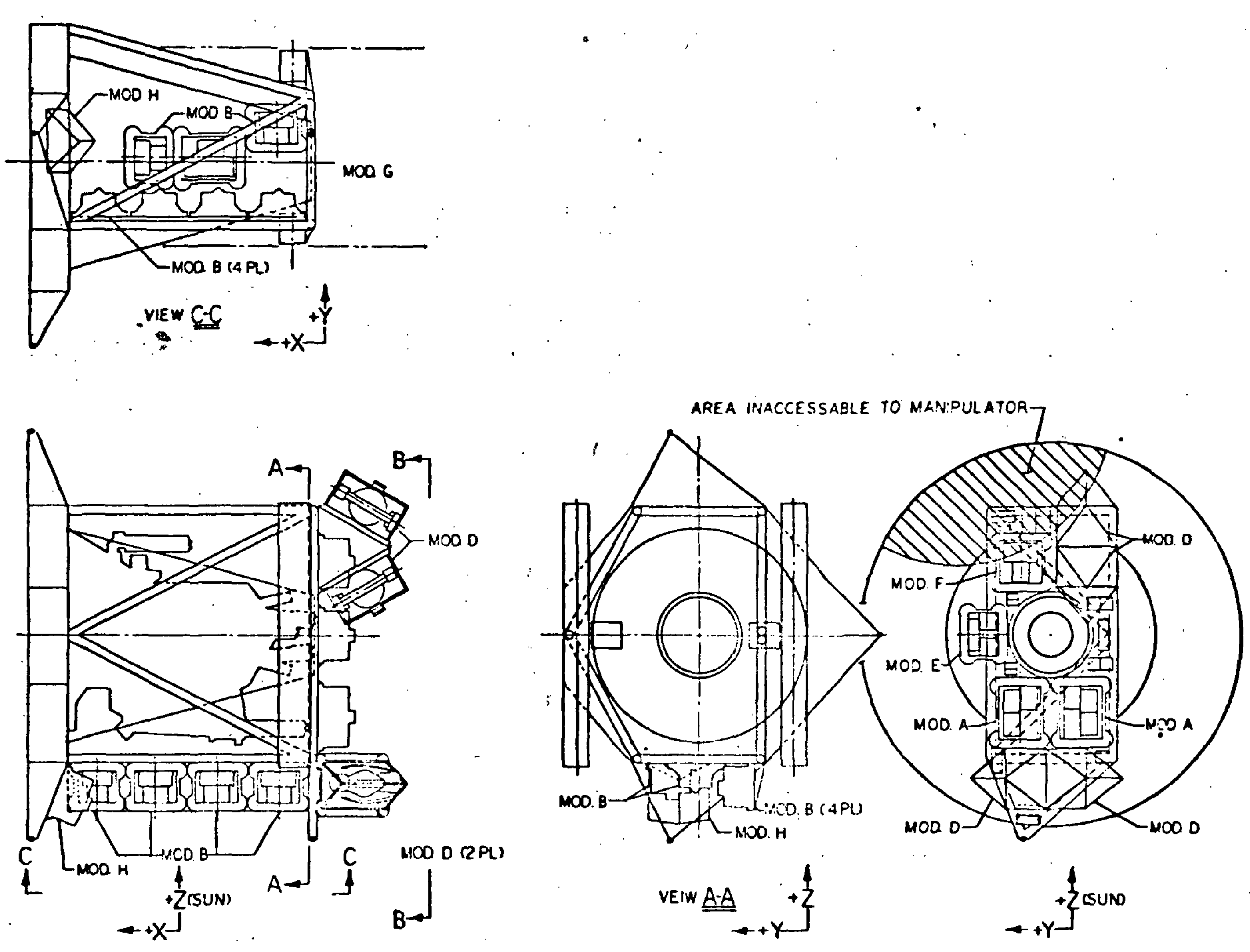


Figure IV-8. Modularized subsystems - tray type - SSM configurations 3B and 6A shown.

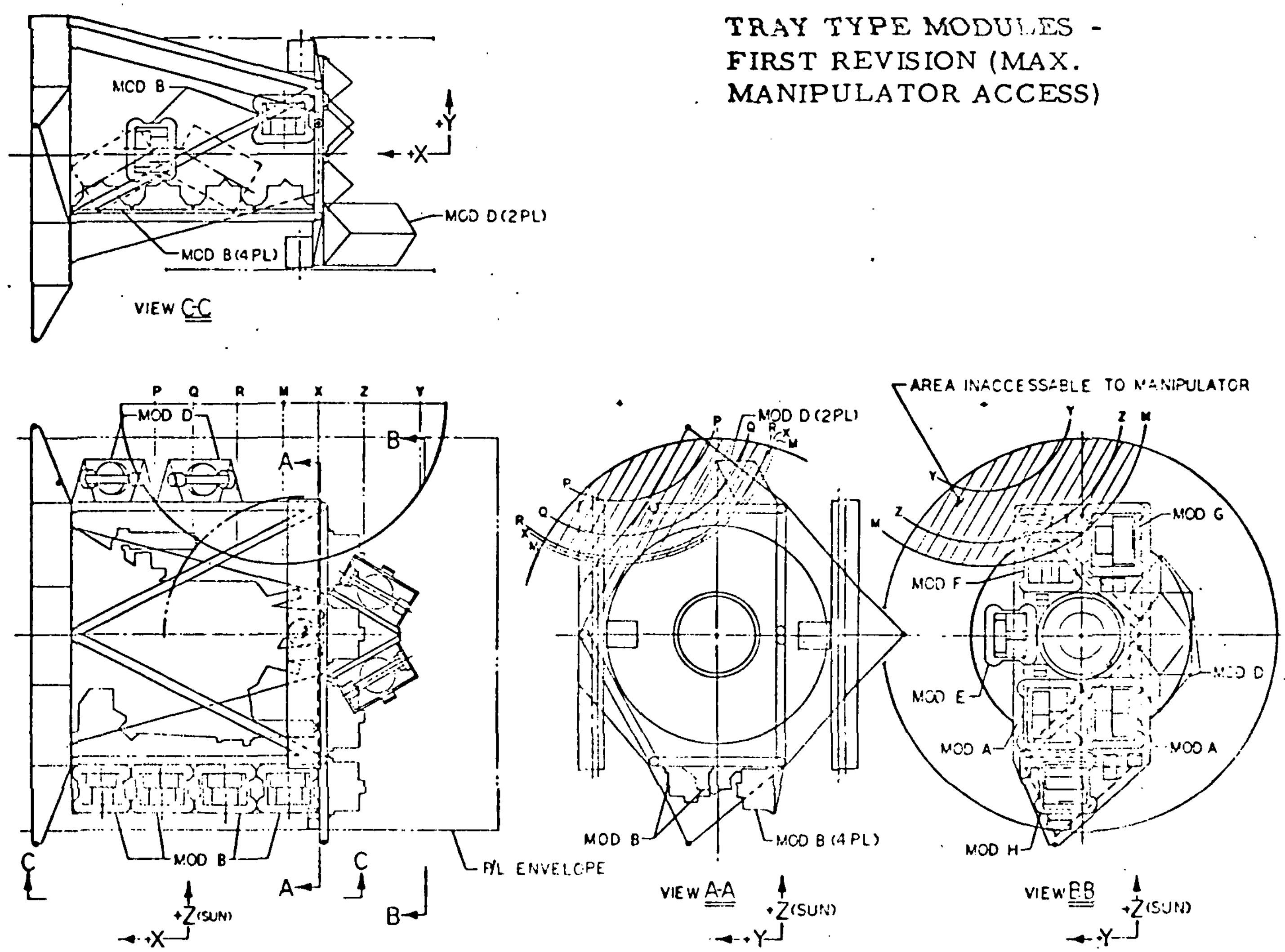


Figure IV-9. LST - modularized subsystem - SSM configurations 3B and 6A shown.

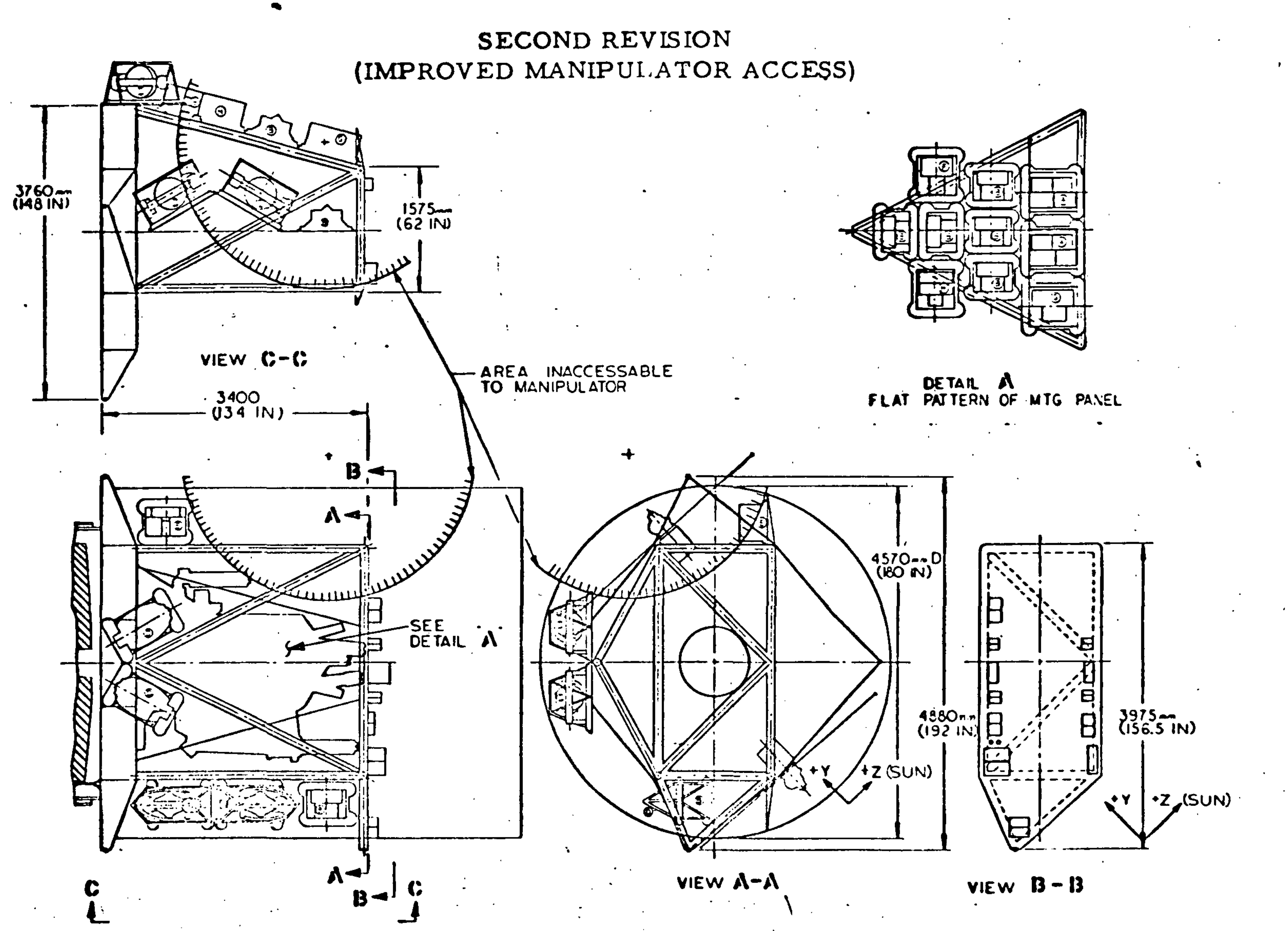


Figure IV-10. Modularized subsystem - SSM configurations 3B and 6A shown.

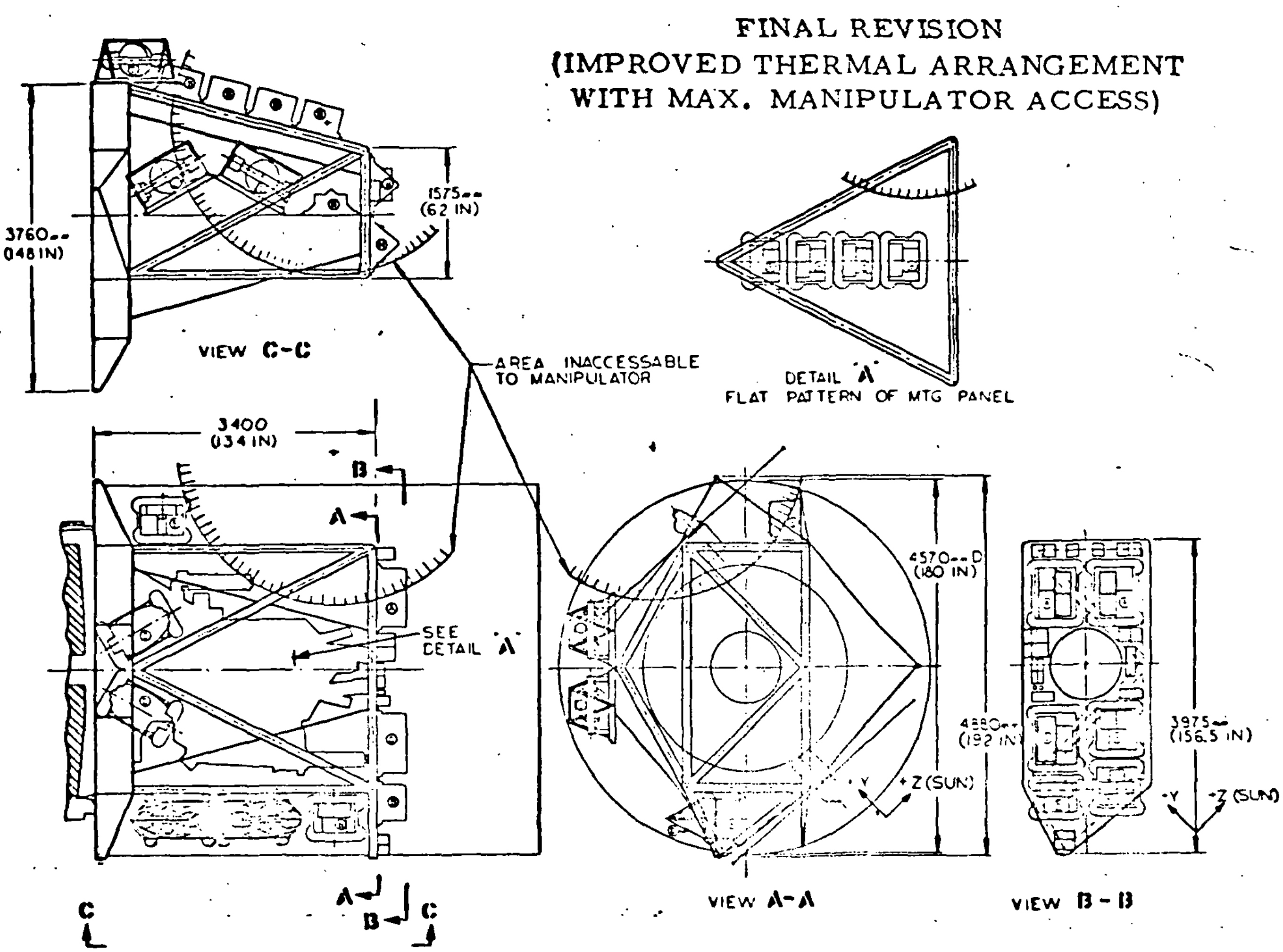


Figure IV-11. LST - modularized subsystem - SSM configurations 3B and 6A shown.

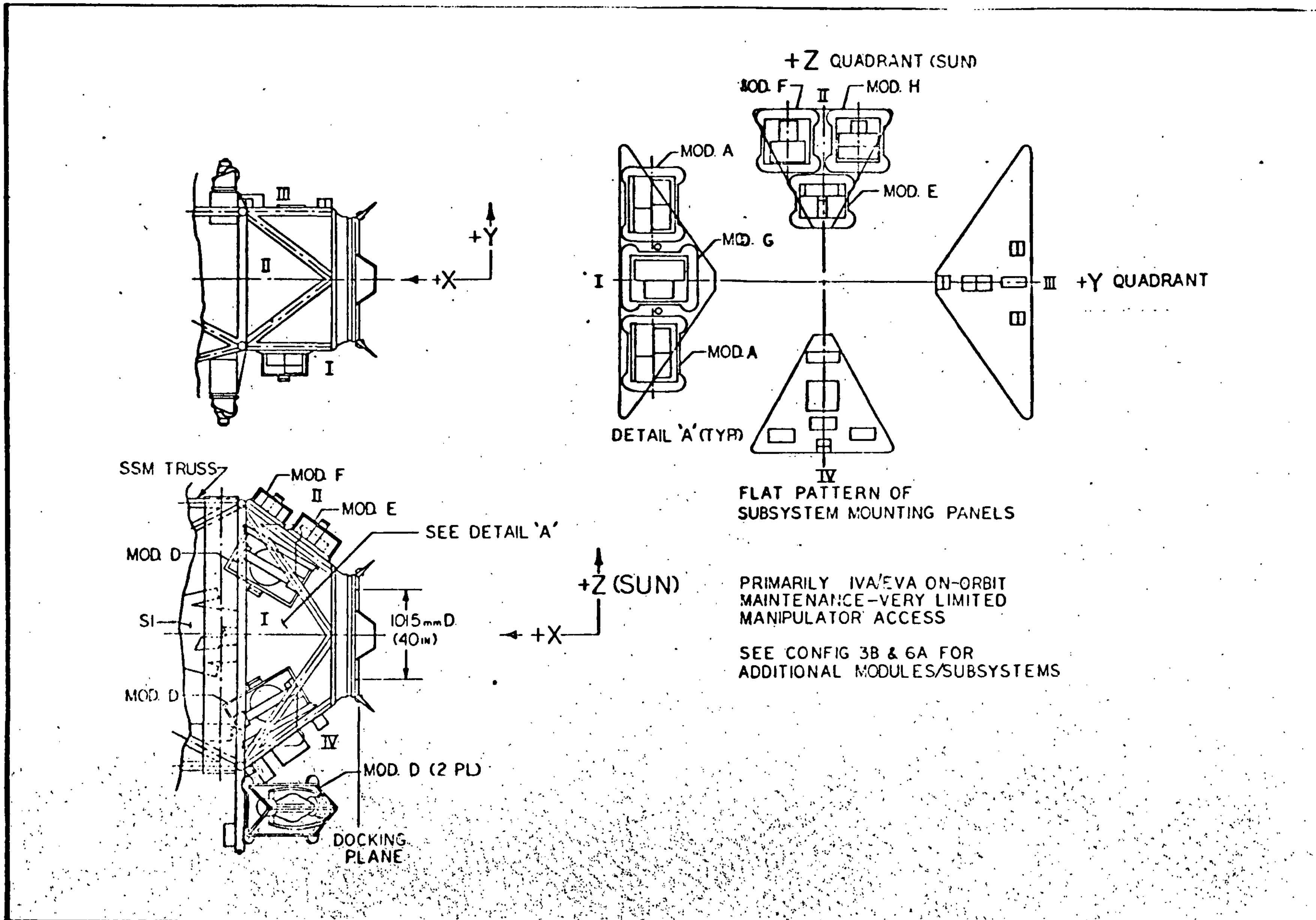


Figure IV-12. LST - modularized subsystems - tray type - SSM configurations 3C and 6B shown.

configuration. The other configurations can easily be modified to match the 1499-mm (59-in.) attachment spacing by reversing the taper of the Primary Frame Assembly or by canting the aft post to get to the 2997-mm (118-in.) spacing required.

D. Unitized Configuration

The unitized SSM could be either a truss or shell, but is distinguished by having the most easily isolatable interfaces between the SI and SSM. In this concept (Fig. IV-13), none of the SSM equipment is packaged around the SI, but is all located aft of the SI. The SSM structure has a splice between the aft-most SI equipment and the forward-most SSM equipment, so that the SSM is separable as a short section containing the equipment, plus a longer adapter. This SSM concept is not intended to be removable as an entity from the OTA/SI on-orbit (although a version of it could be). Rather, it provides the capability of separating the SSM more cleanly from the OTA/SI, both physically and functionally, than the other concepts. These characteristics could be very valuable for more selective separation of the LST major elements for ground testing, or for desensitizing the interfaces between major elements to changes in design of the elements.

E. Shell Configuration

A configuration utilizing trays for mounting subsystems equipment was developed and is shown as Figure IV-14. This configuration could be modified as discussed above to match the latest Shuttle attach points. As noted in Figure IV-14 the CMG and battery modules are inaccessible to the manipulator, hence, the cylindrical shell SSM configuration is least desirable for on-orbit manipulator maintenance. The configuration which was selected as the preferred one for the updated Phase A design is discussed below.

1. General Arrangement. The updated Support Systems Module (SSM) is a cylindrical structure with a total length of 4675 mm (184 in.) and an inside diameter of 3300 mm (130 in.) (Fig. IV-15). The aft end of the SSM is a flat bulkhead with a 1015 mm (40 in.) diameter opening. A removable cover provides micrometeoroid protection while allowing EVA entrance into the SSM. Since the SSM is not pressurized, a pressure seal is not required.

The primary ring and bulkhead on the Phase A design is replaced by the Primary Base Frame Assembly of 380-mm (15-in.) depth which provides

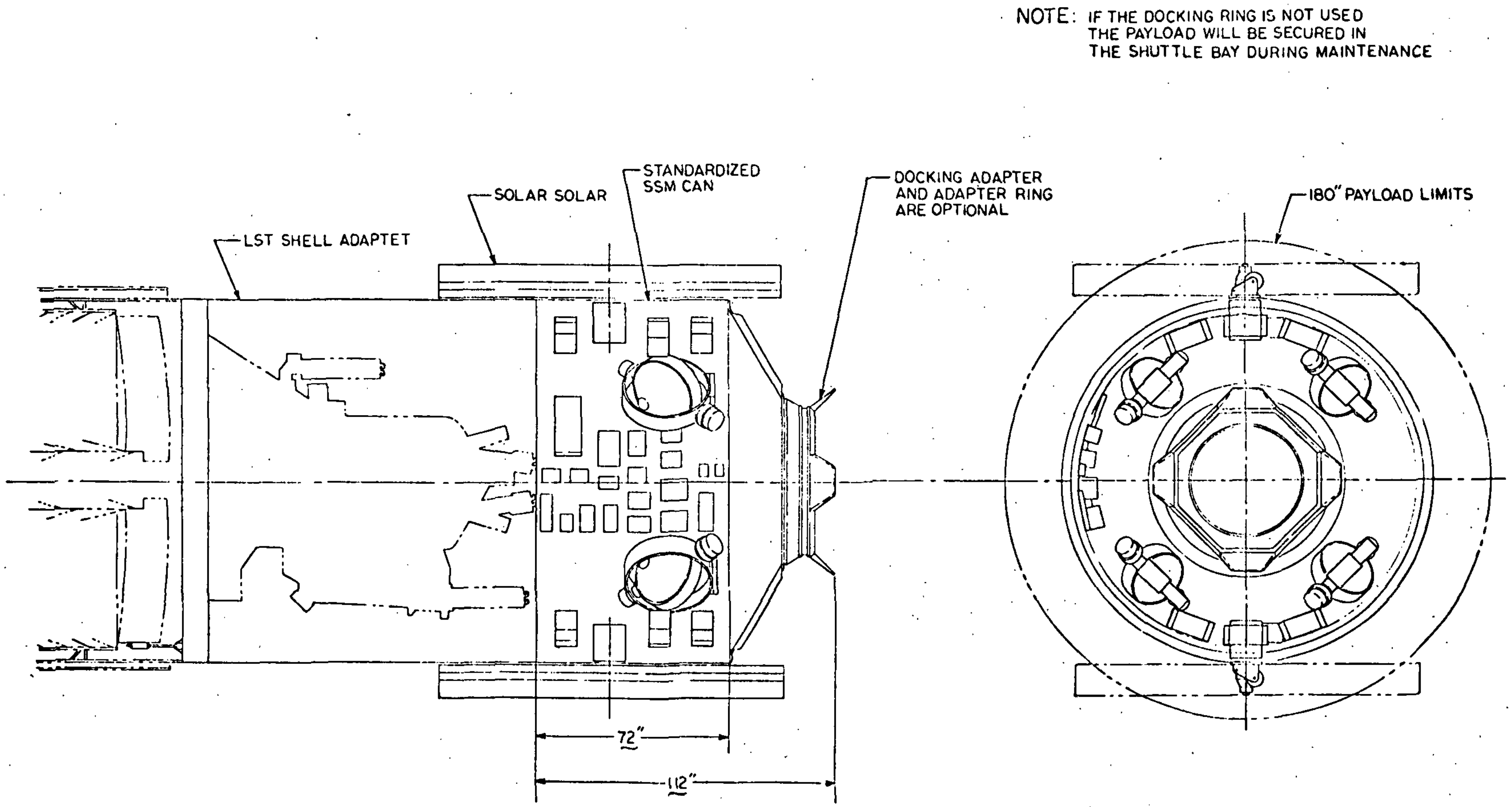


Figure IV-13. Unitized SSM.

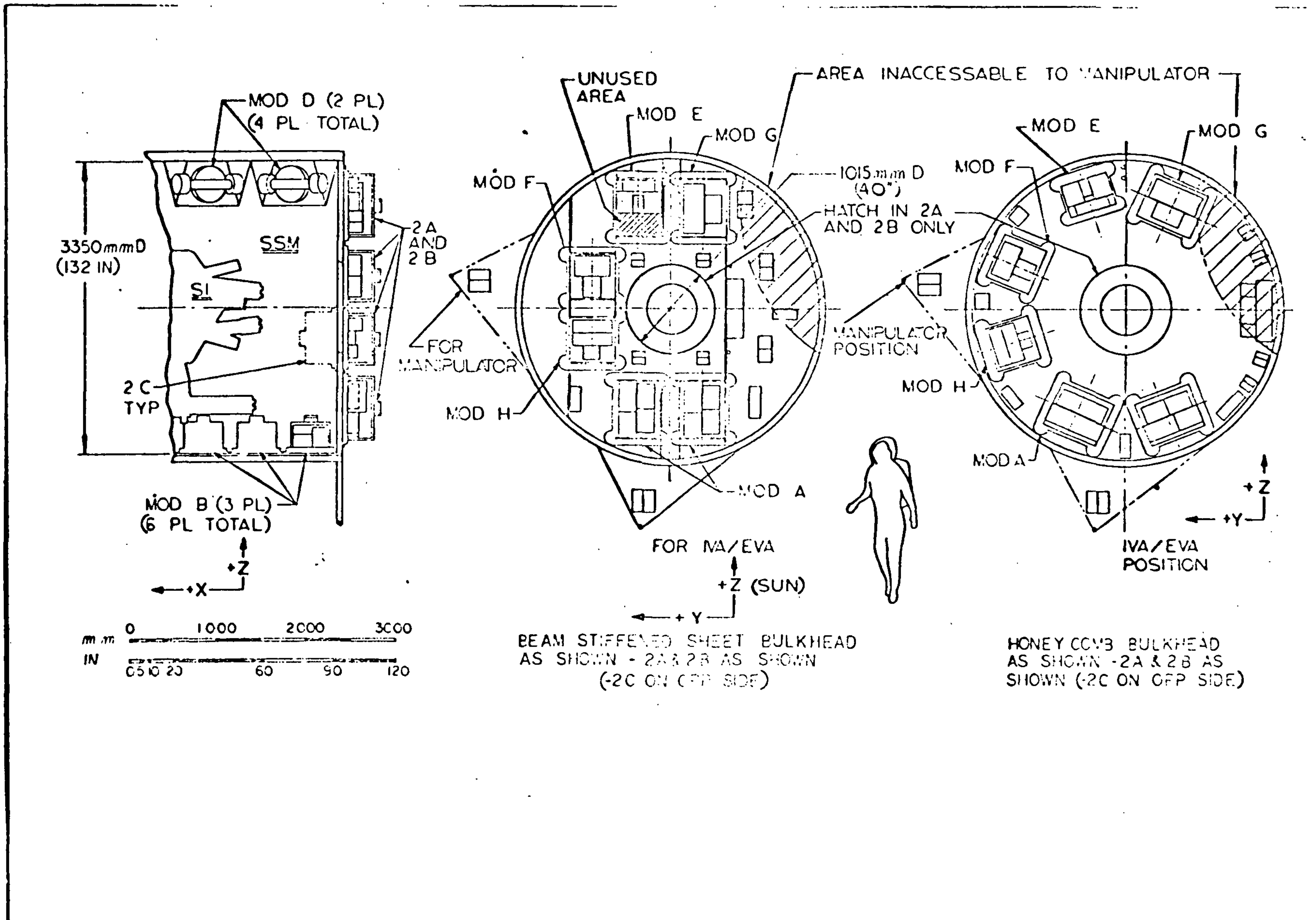


Figure IV-14. LST - modularized subsystems tray type - SSM configurations 2A, 2B and 2C shown.

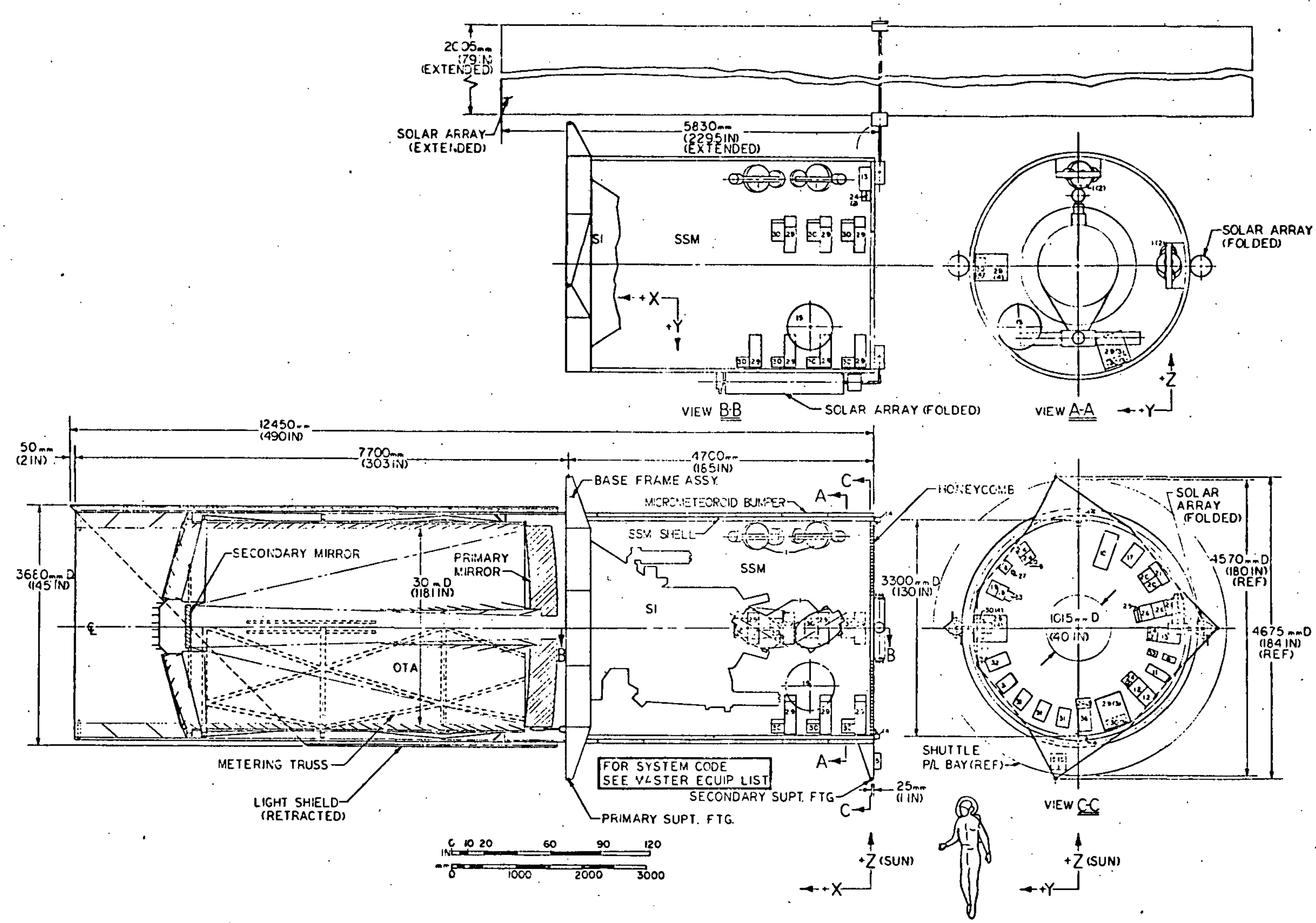


Figure IV-15. Recommended LST configuration (2B modified) long cycle SSM - limited IVA/EVA.

eight points on the forward face for mating with the Optical Telescope Assembly (OTA) truss, eight points on the aft face for mating with the SSM, three points on the forward face for mating with the mirror, and three points on the aft face for mating with the SI. (It is recommended that the SI be changed from the present eight-point mount to a three-point mount.) The OTA meteoroid shield is attached to the primary frame assembly instead of interfacing with the SSM aft of the frame as in the Phase A design.

The Space Shuttle is the only launch vehicle considered in this study. The new Primary Frame Assembly was selected as a potential common structure which may also be used as the chassis of other Shuttle-launched payloads, such as the Earth Observation Satellite (EOS) and the U. S. Domestic Communications Satellite (DOMSAT). The Primary Frame Assembly ties directly to Shuttle-integral fittings at three points. These three points plus a fourth tiepoint on the SSM aft side duplicate the Phase A mounting arrangement, but eliminate the Shuttle-mounted cradle required earlier. The rigid solar array has been replaced by flexible rollup type solar arrays which, when extended, are in approximately the same position as the earlier solar array.

The updated SSM retains the conventional aluminum structure used earlier, except that the lack of a requirement for pressurization may allow thinner, lighter members. Without pressurization, the external SSM meteoroid shield (bumper) is no longer necessary for that purpose. It was retained, however, for its thermal control function, since the thermal control concept was retained. Deletion of the meteoroid shield would require the thermal control multilayer insulation now located external to the shell to be located internally, since thermal control coatings must be applied to the SSM shell's external surface. This would increase the outgassing problem and would also probably cause increased fluctuations in internal equipment temperatures. The deletion of this shield should be investigated further in the Phase B study. Since docking is eliminated (see Section F) the flat aft SSM bulkhead may be made from relatively thin honeycomb. Otherwise the updated SSM construction is essentially the same as described earlier.

2. <u>System Arrangement.</u> The updated systems are arranged as shown in Figure IV-15. Components are coded by number as noted in the Phase A Master Equipment List. The same philosophy as used earlier was used to mount as many components as possible on the aft SSM bulkhead. Again, temperature-critical subsystems were generally placed on the antisun side, whereas thermally inactive items or items with higher allowable temperatures were mounted on the sun side. This time, however, there was more of an attempt to optimize item grouping by function to minimize cabling length

and interfaces, since earlier analyses had indicated that the thermal requirements might not be as severe as previously thought. Batteries and CMGs were located as before. The updated systems arrangement is still not volume balanced although the functional grouping of systems tends this way and, thus, probably improves astronaut access to the systems. In any event, there appears to be room for much more equipment before the packaging density would become a problem. Thermal covers are provided to thermally isolate the SSM systems from the SI. The covers, which are not shown, can be provided either for individual systems, or, if further thermal analysis permits, for clusters of systems more densely packed.

The fixed-head star trackers and the RGA are mounted to the same platform, which is very rigidly mounted to the forward end of the SI as in the Phase A design. Since the SSM is no longer pressurized, the FST shades do not need pressure-tight windows to view; simple cutouts will suffice, unless contamination constraints dictate otherwise.

The new flexible rollup solar array does not block the star trackers in the folded position. The star trackers can now be used without extending the solar array, although the new solar array cannot provide any power in its folded position.

No Contamination Control System (CCS) is shown for the updated configuration. If required, this could be provided during ground maintenance. It is assumed that on-orbit maintenance would not require a CCS.

Although there was not time enough for detailed thermal analyses of the updated system arrangement, experience gleaned from the Phase A configuration indicates that the systems will meet their respective thermal requirements, and that there is adequate EVA clearance for limited astronaut on-orbit maintenance.

Since the Titan is no longer being considered as an alternative launch vehicle, the LST configuration has been allowed to become incompatible with the Titan. If the Titan should ever be reconsidered, the current LST would have to be redesigned.

3. Spacecraft Attachment to the Launch Vehicle. The statically determinate, four-point scheme which was used in the Phase A design to support the spacecraft in the Shuttle was selected for the updated reference configuration. However, use of the Primary Frame Assembly eliminates the need for a ring or cradle, and the attachment fittings are integral with the

Shuttle. As before, the LST structural system is completely relieved from loads induced by structural deflections of the Shuttle during flight. As shown in Figure IV-16, the updated LST is mounted in the Shuttle bay by attachments at Shuttle Station 649 and Station 833 and faces aft. If it becomes necessary to mount the LST facing forward, this can be done by adding a canted post on the opposite side to the existing aft one to pick up attachment points at Shuttle Station 1128 and Station 951. The cant in the post takes the 178-m (7-in.) difference between the spacings.

The updated cylindrical SSM structure is similar enough to the original that the shear post should easily withstand the concentrated attachment interface loads. The Primary Frame Assembly which replaces the OTA main ring and bulkhead is much stiffer and should be more than adequate for launch loads.

4. Equipment Summary and Mass Data. An update to the Phase A Master Equipment List is provided as Table IV-2 for the recommended minor on-orbit maintenance shell-type configuration (Phase A update design).

Mass characteristics for the Phase A update design are provided in Table IV-3.

F. Concept Comparisons

The comparison matrix is shown as Table IV-4. The analysis of these configurations is summarized below. The truss was originally proposed because of advantages in access to equipment which it was felt to have over the shell for a man in a pressure suit or a manipulator. Upon more detailed analysis, however, the access advantages over a suited man appeared more doubtful. Also, as shown in the structural and thermal analyses, there are problems associated with this design which make it less desirable than the shell. In addition, such problems as contamination control and stray light control would probably be intensified with the open truss design. The material mass required for micrometeoroid penetration protection on the truss configuration would be less than that required for the shell design since the truss will have a smaller surface area requiring protection. However, most of the required material mass for protection of the shell would be existing, load-carrying structure and thermal shielding. The mass of material is practically insignificant in either configuration.

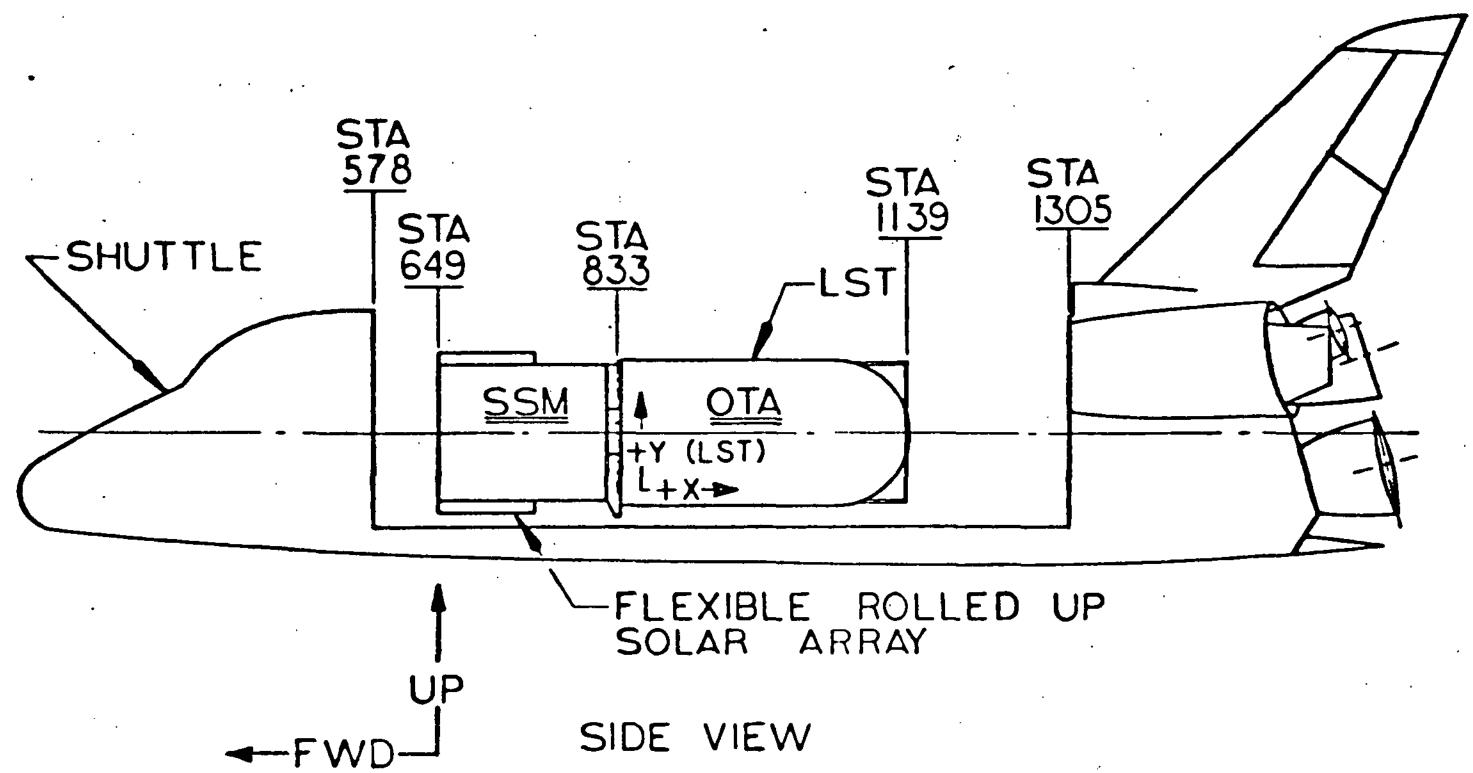


Figure IV-16. LST in Shuttle payload bay - limited IVA/EVA.

TABLE IV-2. CHANGE TO PHASE A LST MASTER EQUIPMENT LIST

COMPONENT	Δ UNITS PRO-VIDED	Δ TOTAL WEIGHT (kg)	Δ TOTAL WEIGHT (lb)	Δ TOTAL AV. PWR (w)	UNIT SIZE (IF CHANGED) (mm)	REMARKS
ATTITUDE CONTROL						
CMG'S & DEA	0	−142.0	−313.0	-28 (−22 SLEW)	813 X 488 X 424	ORIGINAL SIZE DRIVEN BY HEAO COMMONALITY NO LONGER APPLICABLE
COARSE SUN SENSOR	−3	− 0.3	− 0.7	–	–	REDUNDANCY DECREASED FOR COST REDUCTION
RCS ELECTRONICS	−1	− 7.3	− 16.1	–	–	REDUNDANCY DECREASED FOR COST REDUCTION
RCS THRUSTER MOD.	−2	− 3.7	− 8.0	–	–	REDUNDANCY DECREASED FOR COST REDUCTION
RCS LINES, VALVES, REG. SET & MISC. STRUCT.	0	− 7.2	− 15.5	–	–	REDUNDANCY DECREASED FOR COST REDUCTION
RCS TANK	0	+ 20.7	+ 45.7	–	686 DIA.	SKYLAB TANKS USED
GN_2	–	+ 9.1	+ 20.0	–	–	INCREASE DUE TO SUN ORIENTATION REQMT. DURING EMERGENCY 30–DAY HOLD
NET CHANGE		−130.7	−287.6			
ELECTRICAL						
BATTERIES	+1	+100.7	+222.0	–	419 X 508 X 178	ORIGINAL NEW DESIGN CHANGED TO HEAO DESIGN FOR COST REDUCTION (THE QUANTITY INCREASE DUE TO 2½ YR. LIFETIME REQMT.)
CHARGERS	+1	+ 7.0	+ 15.0	+.33	–	SAME AS BATTERIES
REGULATORS	−2	− 3.2	− 7.0	–	305 X 203 X 127	QUANTITY DECREASED FOR COST REDUCTION
SOLAR PWR. DISTR.	−1	− 1.8	− 4.0	–	356 X 203 X 178	QUANTITY DECREASED FOR COST REDUCTION
ELECT. DISTR. UNITS	−4	− 4.5	− 10.0	−10	–	QUANTITY DECREASED FOR COST REDUCTION
SOLAR PANELS	−8	−137.9	−304.0	–	1676 X 5587	CHANGED TO SIMPLER DEPLOYMENT/RETRACTION SYSTEM; RESIZED SMALLER
SOLAR PANEL MECH. & MISC.	0	− 16.4	− 36.2	–	–	CHANGED TO SIMPLER DEPLOYMENT/RETRACTION SYSTEM
ELECT. CONTROL ASSY.	−1	− 7.3	− 16.0	−5	–	QUANTITY DECREASED FOR COST REDUCTION
CABLING & CONNECT.	–	− 22.7	− 50.0	–	–	REVISED ESTIMATE
INACTIVE PREINSTALLED EQUIP. FOR MAINT.						
CAUTION & WARNING CABLE	0	− 0.9	− 2.0	–	–	REVISED ESTIMATE
HEADSET RECEPT.	−3	− 4.5	− 10.0	–	–	NOT REQ'D.
ECLS ELECT. PROV.	–	− 2.3	− 5.0	–	–	NOT REQ'D.
CABLE & TEST ACCESS.	–	− 2.3	− 5.0	–	–	REVISED ESTIMATE
NET CHANGE		− 96.1	−212.2			

TABLE IV-2. (Concluded)

COMPONENT	UNITS PRO-VIDED	Δ TOTAL WEIGHT (kg)	Δ TOTAL WEIGHT (lb)	TOTAL AV. PWR	UNIT SIZE (IF CHANGED) (mm)	REMARKS
COMM. & DATA MGMT.						
TRANSPONDER	0	− 3.6	− 8	+4		APOLLO TRANSPONDER REPLACED WITH A MODIFIED ERTS
CIRCULATOR SWITCH	SEE REMARKS	+ 0.9	+ 2	−	101.6 X 101.6 X 25.4	REPLACES RF SWITCH (2 PROVIDED)
MULTICOUPLER	0	−	−	−	63.5 X 63.5 X 152	OMITTED IN ORIGINAL LIST
DOWNLINK DATA DISTRIBUTOR	0	+ 1.8	+ 4	+4	76.2 X 76.2 X 152	OMITTED IN ORIGINAL LIST
NET CHANGE		− .9	− 2	+8		
THERMAL CONTROL						
NO CHANGE						
CONTAM. CONTROL						
ALL EQUIPT.	−	− 29.1	− 64			NOT REQ'D (PURGE LINE AND DIFFUSER MAY BE REQ'D FOR LAUNCH & RE-ENTRY)
STRUCTURE						
DOCKING PORT EQUIP.		−198	−437			LST LATCHED INTO BAY FOR LIMITED ON-ORBIT MAINTENANCE (COST REDUCTION CHANGE)
SHROUD ADAPTER		−251	−553			NOT REQ'D.
PRIMARY RING		− 51.3	−113			REPLACED BY PRIMARY FRAME
PRESS. BULKHEAD		−177	−391			REPLACED BY PRIMARY FRAME
PRESS. BULKHEAD DOOR		− 48	−106			NOT REQ'D.
AFT CONICAL BULK.		− 98	−216			REPLACED BY AFT PLATE
AFT PLATE		+ 90.8	+200			COST REDUCTION CHANGE
AFT PLATE RING		+ 34	+ 75			COST REDUCTION CHANGE
PRIMARY FRAME & STRUTS		+460	+1015			COST REDUCTION CHANGE
REAR HATCH COVER		+ 10.4	+ 23			OMITTED IN ORIGINAL LIST
SIDELOAD STRUT		+ 11.8	+ 26			COST REDUCTION CHANGE (ELIMINATES ADAPTER)
EQUIPT. MTG. HDW.		+ 68.0	+150			NEW ESTIMATE
HAND HOLDS & FOOT RESTRAINTS		+ 29.9	+ 66			
NET CHANGE		−118.4	−261			
TOTAL NET CHANGE		−377.8	−832.8			

17,989 (PHASE A LST WT. LESS CONTINGENCY) − 833 (Δ WT) = 17,156 + 3,432 (20% CONTINGENCY) = 20,588 (NEW LST WT. IN LBS.)

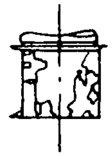
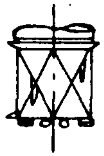
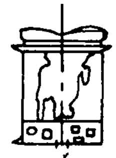
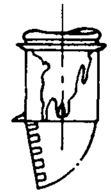
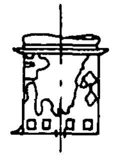
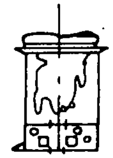
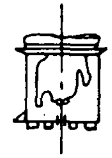
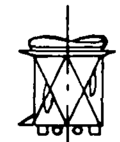
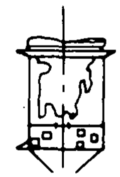
TABLE IV-3. LST PHASE A UPDATE DESIGN MASS CHARACTERISTICS (SOLAR PANEL AND LIGHT SHIELD EXTENDED)

Configuration	Wt kg (lb)	XCG[a] mm (in.)	YCG[a] mm (in.)	ZCG[a] mm (in.)	I_x (kg-m^2) (slug-ft^2)	I_y (kg-m^2) (slug-ft^2)	I_z (kg-m^2) (slug-ft^2)
Minor On-Orbit Maint (Shell Config with 20 Percent Contingency)	9 339 (20 588)	4602 (181.2)	7.6 (0.3)	-38.1 (-1.5)	18 777 (13 849)	99 632 (73 485)	104 709 (77 229)

a. The reference axes used in the Phase A study were used here for ease in comparison to that data, even though the origin has been shifted in the Phase B study guidelines. The origin here is the aft surface of the aft plate.

NOTE: This is the final update of the mass characteristics data in Table IV-6 for the selected configuration. The data in Table IV-6 were utilized for attitude control system sizing in the study.

TABLE IV-4. LST PHASE A DESIGN UPDATE CONCEPTS

CRITERIA	"PURE" GROUND MAINTENANCE CONCEPTS			"MINOR" ON-ORBIT MAINTENANCE CONCEPTS					
				EVA			SHUTTLE MANIPULATOR AND/OR EVA		
	 SHORT SHELL	 SHORT TRUSS	 UNITIZED SSM (SHORT ADAPTER)	 SHORT SHELL (HINGED DOOR)	 LONG SHELL (INTERNAL COMPONENTS)	 UNITIZED SSM (LONG ADAPTER)	 LONG SHELL (EXTERNAL COMPONENTS)	 SHORT TRUSS	 UNITIZED SSM (LONG ADAPTER, HINGED DOORS)
1. LST kg (lb) (incl 20 percent conting)	9 147 (20 166)	9 233 (20 354)	9 437 (20 804)	9 172 (20 221)	9 339 (20 588)	9 585 (21 132)	9 570 (21 097)	9 646 (21 266)	9 833 (21 678)
2. LST m (ft)	11.1 (36.3)	11.1 (36.3)	12.5 (41.1)	11.1 (36.3)	12.4 (40.8)	13.4 (44.1)	13.2 (43.3)	11.9 (38.9)	13.4 (44.1)
3. Maint level	← Black box or higher →			← Black box[a] →			← Multibox trays[a] →		
4. Spares & logistics wt & vol to orbit	N/A	N/A	N/A	← See Table VI-1 →			Table VI-1 Plus approx 193 kg (425 lb) for trays.		

a. Science instruments replaced by man in all concepts.

TABLE IV-4. (Continued)

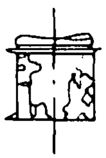
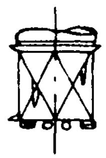
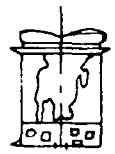
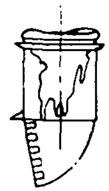
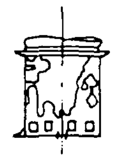
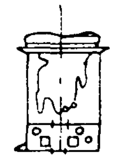
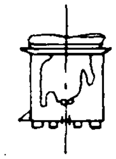
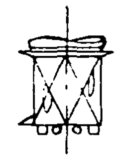
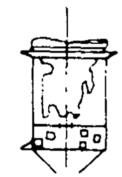
<table>
<tr><th rowspan="3">CRITERIA</th><th colspan="3">"PURE" GROUND MAINTENANCE CONCEPTS</th><th colspan="6">"MINOR" ON-ORBIT MAINTENANCE CONCEPTS</th></tr>
<tr><th colspan="3"></th><th colspan="3">EVA</th><th colspan="3">SHUTTLE MANIPULATOR AND/OR EVA</th></tr>
<tr><th> SHORT SHELL</th><th> SHORT TRUSS</th><th> UNITIZED SSM (SHORT ADAPTER)</th><th> SHORT SHELL (HINGED DOOR)</th><th> LONG SHELL (INTERNAL COMPONENTS)</th><th> UNITIZED SSM (LONG ADAPTER)</th><th> LONG SHELL (EXTERNAL COMPONENTS)</th><th> SHORT TRUSS</th><th> UNITIZED SSM (LONG ADAPTER HINGED DOORS)</th></tr>
<tr><td>5. No. of flights req.</td><td colspan="3">2 Up, 1 Down</td><td colspan="6">1 Up, 1 Down</td></tr>
<tr><td>6. Qty/ complexity of maint. ops</td><td colspan="9">See timelines, Chapter III</td></tr>
<tr><td>7. Down-time for science</td><td colspan="9">See timelines, Chapter III</td></tr>
<tr><td>8. Crew hazard</td><td colspan="3">None</td><td colspan="3">Near minimum</td><td colspan="3">Minimum</td></tr>
<tr><td>9. Conting capability</td><td colspan="3">N/A</td><td colspan="6">Unplanned on-orbit operations or earth-return</td></tr>
</table>

TABLE IV-4. (Continued)

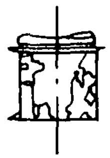
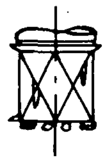
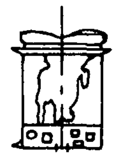
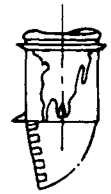
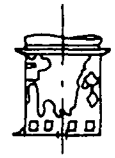
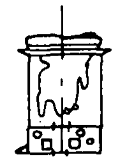
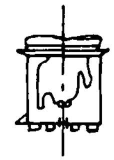
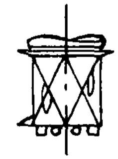
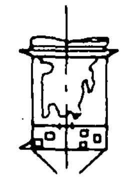
CRITERIA	"PURE" GROUND MAINTENANCE CONCEPTS			"MINOR" ON-ORBIT MAINTENANCE CONCEPTS					
				EVA			SHUTTLE MANIPULATOR AND/OR EVA		
	 SHORT SHELL	 SHORT TRUSS	 UNITIZED SSM (SHORT ADAPTER)	 SHORT SHELL (HINGED DOOR)	 LONG SHELL (INTERNAL COMPONENTS)	 UNITIZED SSM (LONG ADAPTER)	 LONG SHELL (EXTERNAL COMPONENTS)	 SHORT TRUSS	 UNITIZED SSM (LONG ADAPTER, HINGED DOORS)
10. Constraints imposed on Shuttle	None			3.46 m (11.3 ft) clearance for door to open	None	None	2 m (6.5 ft) clearance for component removal	2 m (6.5 ft) clearance for component removal	1.73 m (5.7 ft) clearance for door to open
11. Ground facilities req'd	See Chapter III			None for on-orbit maint. — same as pure-earth return concept for major overhaul (approx each 5 yr.)					
12. Support equipt	See Chapter III								
13. Docking HDW required	No								

TABLE IV-4. (Continued)

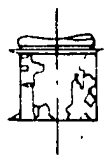
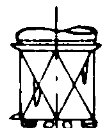
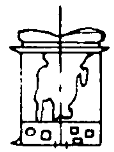
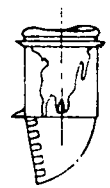
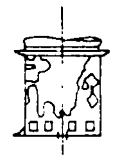
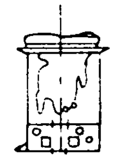
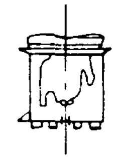
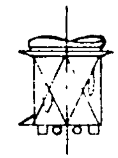
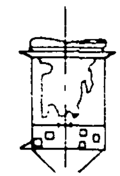
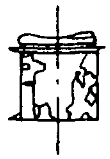
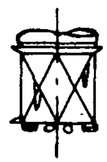
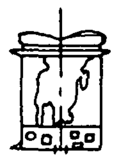
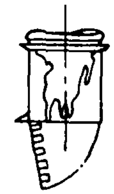
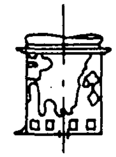
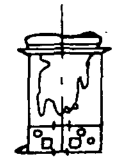
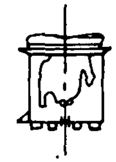
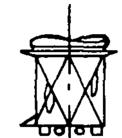
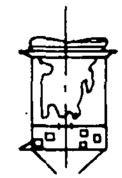
CRITERIA	"PURE" GROUND MAINTENANCE CONCEPTS			"MINOR" ON-ORBIT MAINTENANCE CONCEPTS					
				EVA			SHUTTLE MANIPULATOR AND/OR EVA		
	 SHORT SHELL	 SHORT TRUSS	 UNITIZED SSM (SHORT ADAPTER)	 SHORT SHELL (HINGED DOOR)	 LONG SHELL (INTERNAL COMPONENTS)	 UNITIZED SSM (LONG ADAPTER)	 LONG SHELL (EXTERNAL COMPONENTS)	 SHORT TRUSS	 UNITIZED SSM (LONG ADAPTER, HINGED DOORS)
14. Docking HDW option available	← Yes →			← No →	← Yes →				← No →
15. Growth potential & flexibility to design changes	Poor	Poor	Fair	Poor	Good	Best	Fair	Good	Fair
16. Interconnection qty & complexity	← Moderate →						← A fairly major problem for manipulators →		
17. Access	Fair	Best	Good	Good	Good	Good	Good	Best	Good
							← Requires manipulator to roll LST 180° in bay →		

TABLE IV-4. (Concluded)

CRITERIA	"PURE" GROUND MAINTENANCE CONCEPTS			"MINOR" ON-ORBIT MAINTENANCE CONCEPTS					
				EVA			SHUTTLE MANIPULATOR AND/OR EVA		
	 SHORT SHELL	 SHORT TRUSS	 UNITIZED SSM (SHORT ADAPTER)	 SHORT SHELL (HINGED DOOR)	 LONG SHELL (INTERNAL COMPONENTS)	 UNITIZED SSM (LONG ADAPTER)	 LONG SHELL (EXTERNAL COMPONENTS)	 SHORT TRUSS	 UNITIZED SSM (LONG ADAPTER, HINGED DOORS)
18. Contam. protection	Good	Poor	Best	Good	Good	Best	Good	Poor	Best
19. Adaptability to move extensive on-orbit maint. later	← Good →			← Better →					

The access advantage that the open truss was originally felt to have is not presently needed for suited maintenance, since the shell provides ample access room, including room for growth. Hence, the truss was dropped from EVA maintenance consideration. It was still considered, however, as a candidate for manipulator maintenance.

The SI equipment, by its nature and location, does not lend itself easily to manipulator maintenance approaches, and hence would be replaced by a suited astronaut in any of the configurations. Since the instrument sensors are the most likely of any of the LST equipment to fail or be degraded (the RADC Reliability Notebook indicates a failure rate of 130 failures per million hours for a vidicon, compared to about 15 failures per million hours maximum assumed for other LST equipment), some or all of the instrument sensors (cameras) are candidates for replacement in the minor maintenance operation on-orbit. Since this operation must be a suited one, it was determined that the limited amount of other equipment to be replaced on-orbit should also be done in a suited mode rather than by manipulator, and hence the truss was dropped from further consideration.

Since the equipment to be replaced is limited, it was determined that the components (black boxes) should be mounted individually rather than in trays. The LST design will not be impacted significantly by suited mode operation. Skylab and Apollo EVA experience and human factors designs are available for use on LST. The main considerations in mounting the equipment for suited operations are to provide access to the components, sufficient space between components for use of a gloved hand to disconnect cables, quick-disconnect mounting bolts, provide sufficient handholds and foot restraints, and adequate lighting.

In comparing the unitized SSM concept with the others, it seemed that the Phase A design effectively provides the same thermal isolation between SI and SSM components which would be available with the unitized SSM. In the nonunitized SSMs, the few SSM components which are mounted around the SI (as well as all other SSM components) are isolated thermally by insulation from the science instruments and other equipment. Each SSM component radiates only to the small area of structure to which it is mounted. The science instrument detectors radiate primarily to the SSM wall adjacent to them, but also can radiate to the portions of the SSM aft of them. This does not interfere with the SSM components and should permit greater radiative growth area for these detectors. The unitized SSM provides thermal isolation by a thin insulated bulkhead (with access hatch) between the SI and SSM component section. The SSM components could possibly be left

uninsulated in this configuration, which would be an advantage in access and cost. No significant advantage could be determined in having the field splice just aft of the SI which was provided as part of the unitized SSM; and the splice complicates the structure somewhat, which is a disadvantage. The unitized SSM also increases the length of the LST slightly which is a disadvantage, although not a drastic one. During the study, there was not sufficient data to determine which of these approaches is best, and since the nature of the SI will probably change considerably during the Phase B study, the decision was made to remain with the nonunitized concept. Other factors could become more dominant as the study progresses, such as isolation of the SSM components and wiring from the SI instruments for contamination control (contamination caused by traffic during ground operations or outgassing on-orbit), or more selective separability of the LST elements for ground testing. It should not be a drastic impact to change to a unitized design from the present concept, if this is desired in the future.

Docking capability could be added to most of the concepts, but was specifically omitted as a cost reduction item. With the current design, the LST can be latched back into the bay for maintenance, using the same structural mounting arrangement required for launch, and an EVA maintenance operation can be conducted just as easily as if it were docked to the Shuttle docking module. The LST is presently designed for launch with SSM forward. That is also the orientation required for on-orbit maintenance. Because of c.g. constraints, the LST may have to be launched with the SSM aft. Even in that case, however, the SSM-forward orientation is still required for on-orbit maintenance. Because of some irregular spacing of support points in the forward part of the payload bay, a slight redesign of the aft support strut on the SSM would be required in that case to adapt the SSM to two different support-point spacings. This approach eliminates having to carry the docking module, which saves approximately 1361 kg (3000 lb), 2.2-m (7 1/2-ft) length, and the 198-kg (437-lb) weight and cost of docking hardware on the LST. Initially some concern was felt that docking capability should be provided for any on-orbit maintenance operation to eliminate the concern of not being able to insert the LST into the bay because of failed retraction mechanism on the light shield or solar arrays. However, very reliable designs must be utilized for all retraction mechanisms regardless of the maintenance operation question. Otherwise, the LST can neither be returned to earth for the primary maintenance operation nor installed in the bay for the limited on-orbit maintenance. Manual backup and/or jettison capability must be provided for these mechanisms to ensure earth-return capability. A further backup to the retraction mechanisms of providing a docking capability to be utilized for only the limited on-orbit maintenance does not seem justified.

The hinged rear plate concept with the short shell permits access by the manipulator to internally mounted equipment and to all instruments which would not otherwise be possible with the short shell. For suited operation, a slight increase in shell length to allow access would be simpler and cheaper than the more complex hinged plate concept. Hence, the hinged plate was not considered further.

In all concepts, the components could be mounted externally or internally, particularly on the aft plate. For suited operation, either is acceptable, and since the internal mounting offers more protection from the environment, it was selected.

Any of the concepts will allow a greater degree of on-orbit maintenance to be performed later, if desired. In the recommended configuration (shell with components mounted individually), the main impact of increasing the degree of maintenance is the on-orbit time required. By later packaging some of the equipment on trays or pallets, maintenance time could be reduced, and yet a greater degree of maintenance could be performed.

A list showing delta weights and quantity of equipment between the Phase A update design and the pure ground maintenance short shell configuration is provided as Table IV-5. Mass characteristics are provided in Table IV-6 for the pure ground return maintenance configuration and the minor on-orbit maintenance configuration. As pointed out earlier, the difference between these concepts is very minor and indicates that the increased program flexibility of having minor on-orbit maintenance capability has only a small impact on design.

TABLE IV-5. PURE GROUND MAINTENANCE CONCEPT (SHORT SHELL) WEIGHTS (QUANTITIES ARE Δ's FROM TABLE IV-2 LISTING)

Item	Δ Wt. kg (lb)		Δ Pwr (W)
Lighting	-9.1	(-20)	-30
Handholds and foot restraints	-29.9	(-66)	-
Struct. sidewall (-1.3 m)	-107.5	(-237)	-
Thermal control sidewall (-1.3 m)	-12.7	(-28)	-
TOTAL	-159.2	(-351)	-30

17 156 (minor on-orbit EVA maint config) - 351 (pure ground-return maint config) = 16 805 + 20 percent = 20 166 lb.

TABLE IV-6. LST PHASE A UPDATE DESIGN MASS CHARACTERISTICS (SOLAR PANELS AND LIGHT SHIELD EXTENDED)

Configuration	Wt kg (lb)	XCG[a] mm (in.)	YCG[a] mm (in.)	ZCG[a] mm (in.)	I_x kg-m^2 (slug-ft^2)	I_y kg-m^2 (slug-ft^2)	I_z kg-m^2 (slug-ft^2)
Pure ground ret maint (shell config with no contingency)	7 294 (16 080)	3 759 (148)	17.8 (0.7057)	-2.5 (-0.0969)	16 338 (12 050)	77 306 (57 018)	82 289 (60 693)
Minor on-orbit maint (shell config with no contingency)	7 384 (16 279)	4 420 (174)	-17.9 (-0.7047)	-27.9 (-1.0995)	16 687 (12 308)	92 483 (68 212)	97 471 (71 891)
Pure ground ret maint (shell config with 20-percent contingency)	8 753 (19 296)	3 937 (155)	14.9 (0.5881)	-2.0 (-0.0807)	17 429 (12 855)	79 349 (58 525)	84 333 (62 201)
Minor on-orbit maint (shell config with 20-percent contingency)	8 861 (19 535)	4 572 (180)	-14.9 (-0.5873)	-23.3 (-0.9162)	17 794 (13 124)	94 225 (69 497)	99 213 (73 176)

a. The reference axes used in the Phase A study were used here for ease in comparison to that data, even though the origin has been shifted in the Phase B study guidelines. The origin here is the aft surface of the aft plate.

CHAPTER V. INTERFACES

The key areas where interfaces have changed since the Phase A report are given below.

A. LST-to-Launch Vehicle

1. Structural. Neither the launch support cradle (launch and return modes) nor the docking module (limited on-orbit maintenance mode) is required for mounting the LST to the Shuttle since the new mounting structures are part of the LST and remain with it in orbit. The length of the LST has been selected such that the mounting struts interface with the Shuttle slots in the most efficient manner. Should the LST length (or Shuttle slot spacing) change, relocation of the struts would be required, with a local strengthening of the LST structure. It appears that the LST may have to be launched and returned with SSM aft (because of Orbiter c.g. envelope constraints) and that it should be oriented with the SSM forward during on-orbit maintenance. With the Shuttle bay mounting slots evenly spaced at 1499 mm (59 in.) [a] this should be no problem. There is a potential safety hazard attendant to launching or returning the LST with the SSM aft since in the event of a crash landing there is very little primary structure on the LST between the mirror and the cabin to stop the mirror should it break loose from its mounts. With the SSM forward, this hazard is significantly reduced.

2. Center of Gravity Envelope. The LST center of gravity is estimated to be about 6198 mm (244 in.) from the forward bulkhead (see Fig. IV-16 and Table IV-4) in the launch or return configuration with SSM forward (assumes 8-inch c.g. shift from the X-axis location in Table IV-4 because of retraction of light shield and solar arrays).

3. Docking Hardware. There is no docking hardware required.

4. EC/LSS. This is required only for astronauts in cabin and during EVA; there is no LST pressurizable area to control.

5. Thermal. There is no LST equipment heat dissipation required.

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a. Expected change to Space Shuttle System Payloads Accommodations. JSC 07700, Vol. 14, April 13, 1973.

6. Power. The C &W is reduced to TBD watts; power is required for two pressure suits. Contamination control power is eliminated (874 W average, 1250 W peak, 61 kW-hr).

7. Communications. The portable transceiver and cabling are eliminated in the LST; communications are accomplished by means of the suit.

8. Maintainability. Tools and spares are stowed in a sealed container in the bay and suits and consumables are stored in the pressurized area of the cabin (See Table VI-1). The spares and maintenance support equipment listed in Tables IV-5, IV-6 and IV-7 of Volume II of the Phase A LST Report are no longer applicable.

B. SSM/OTA/SI Interfaces

1. Structural. The main ring, pressure bulkhead, and door have been replaced by the primary frame truss. The pressure seal has been eliminated. The SI interface with the OTA will be three points rather than the eight points defined in the Phase A study; the SSM interface with the OTA primary frame assembly is at eight points instead of a continuous ring. The OTA meteoroid shield ties into the primary frame instead of interfacing with the SSM aft of the frame.

2. Contamination Control. The ducts, filters, and SI cover for contamination control have been eliminated. Later some type of cover may have to be supplied for the SI region to protect the sensitive instrument faces during normal operations on the ground and to provide a region of net positive delta pressure for the SI during launch and reentry purges, if these are required.

3. Thermal. There is no insulation barrier mounted on the primary flat plate truss to isolate the primary mirror from the SI area, as it is assumed that the mirror has insulation mounted on its aft surface as part of the electrical heater system for it.

CHAPTER VI. MAINTENANCE/MAINTAINABILITY

A. Spares/Logistics Quantities

In a typical maintenance operation (at the end of the first 2 1/2 years, for example) some of the LST equipment would be near the end of its expected lifetime, and it would be time to update other equipment (especially science instruments would need to be updated) with more recently developed equipment. Still other equipment would have randomly failed and would need replacement.

The list of equipment in Table VIII-9 of Volume V of the Phase A report was used as a guide to determine the SSM items required to be replaced in typical maintenance operations. The columns dealing with the lowest renewal probability list only the life-limited items (batteries and tape recorders), which should be considered the lower bound of the SSM spares required for minor maintenance. The columns dealing with the middle-range renewal probability should approximate the upper bound of the SSM spares required for minor maintenance. (The moderate maintenance spares quantity should fall between the middle columns and the right-hand columns of Table VIII-9.) The OTA and SI spares discussed in a later paragraph must be added to the list of SSM spares. In addition to replacement of the aforementioned types of equipment, major overhaul would include such things as renewal of the thermal control surface, repair of micrometeoroid damage, recoating of the optical elements, replacement of solar cells, teardown and recleaning of the entire LST, and reverification of all systems in the operational environment.

It is expected that earth-return maintenance would fall into the category of a mjaor overhaul for several reasons. One reason is that the lifetime of the solar panels, thermal coating, etc., would probably be designed for only the period between expected returns to earth and, hence, would normally need rework at each return. Second, the return to earth is likely to be the most contamination-producing event that the LST will experience, because of the difficulty of sealing the LST from the dirtier cargo bay after it is retrieved into the bay, and because of the potential influx of hot gases and contaminants into the bay upon reentry. This contamination will probably necessitate a partial teardown and recleaning of the LST before relaunch. Third, there will be the tendency to do more things to the LST on the ground because of its availability than might be absolutely required.

An attempt was made to quantify the minor on-orbit maintenance spares required. For the SSM, the list of minor maintenance spares mentioned above was utilized. The list of science and telescope instrument sensors to be replaced must be added to the list of minor maintenance SSM spares. These appear to be the most life-limited of all the LST equipment (failure rate of a vidicon is 130 per million hours in the RADC Reliability Notebook compared to approximately 15 per million hours maximum used for other LST equipment).

A summary of the most likely spares required for the minor on-orbit maintenance mode is provided in Table VI-1, in descending order of priority. The priority is a best estimate and is based partially on the instrument rankings listed in the Phase B Statement of Work Guidelines and partially on the estimated instrument utilization times (Table 10-1 of Volume IV of the Phase A Report). This list is more nearly a shopping list than a typical maintenance load, but it has been treated as the latter in this study to assess the impact. Telescope instruments have not been included herein (except for the fine guidance assembly), but such items as the figure, focus, decenter, and angle sensors will probably have to be considered as high-failure items. It is possible that future concepts of the fine guidance assembly will have sensors which are individually replaceable, possibly saving some spares weight and volume, but adding operations time for replacement. Table VI-1 also lists the estimated time required to perform suited maintenance, and the EVA consumables required.

At some degree of EVA on-orbit maintenance beyond minor maintenance, maintenance time becomes dominant and may necessitate grouping of equipment on trays or some similar mounting device, for greater replacement efficiency. Layouts of such arrangements are shown in Chapter IV. The equipment in these layouts represents a moderate on-orbit maintenance level, as depicted in the right-hand columns of Table VIII-9, Volume V, of the Phase A Report.

The spares and tool kit (see Table VI-1) will be mounted in the bay in a compartment which is sealed against contamination. This will eliminate the frequent trips through the airlock by the astronaut to get spares, which would be required if they were stored in the cabin.

For on-orbit limited maintenance, the LST will be placed into the bay by the Shuttle manipulator, with the SSM facing forward. Since the bay doors will probably remain open during servicing, all LST maintenance is considered EVA. The launch locks will be utilized for holding it here while a suited

TABLE VI-1. MINOR ON-ORBIT MAINTENANCE SPARES AND LOGISTICS[a]

Item	Qty	Total Wt.[f] kg	(lb)	Total Vol[g] m^3	(ft^3)	Total EVA Maint. Time Req'd (hr)
Batteries	7	235	(518)	0.397	(14.03)	3.5
Tape recorders	3	19.6	(43.2)	0.036	(1.26)	1.0
Fine guidance assembly	1	139	(308)	0.145	(5.12)	1.0
f/96 camera	3	241.7	(532.8)	0.102	(3.6)	1.0
Faint obj. spectr. (FOS1)[b]	1	67.0	(147.6)	0.15	(5.25)	1.0
High res Echelle spectr (HRS1)[b]	1	62.0	(136.8)	0.242	(8.55)	1.0
High res Echelle spectr (HRS2)[b]	1	62.0	(136.8)	0.22	(7.8)	1.0
Faint obj spectr (FOS2)[b]	1	54.4	(120)	0.276	(9.75)	1.0
Slit jaw camera	1	50.6	(111.6)	0.41	(14.55)	1.0
Faint obj spectr (FOS3)[b]	1	53.3	(117.6)	0.111	(3.9)	1.0
f/12 camera	1	81.6	(180)	0.048	(1.71)	1.0
Faint obj spectr (FOS4)[b]	1	21.7	(48)	0.15	(5.25)	1.0
RGA	1	12.5	(27.6)	0.017	(0.6)	0.7
DPA	1	7.6	(16.8)	0.009	(0.3)	0.7
CMG	1	96.8	(214)	0.69	(24.5)	1.0
Regulator	1	4.3	(9.6)	0.012	(0.45)	1.0
Remote decoder	1	0.6	(1.2)	0.005	(0.15)	0.8
DAU	1	0.6	(1.2)	0.0006	(0.0075)	1.0
Spares subtotal		1210	(2670)	3.02	(106.8)	19.7
EVA tool kit		16.3	(36)	0.13	(4.5)	
Cargo bay subtotal		1226.3	(2706)	3.15	(111.3)	
Two suits		90.7	(200)	0.33	(11.8)	
EVA consumables[c,d,e]		170.0	(375)	2.0	(72.0)	
Cabin subtotal		260.7	(575)	2.33	(83.8)	
Totals		1487	(3281)	5.48	(195.1)	

a. Listed in decreasing order of expected replacement need (determined by reliability, operating hours, and ranking of importance).

b. Ref. Table 10-1 of Vol. IV of Phase A Report for abbreviations.

c. The allowance for H_2O and associated tankage will be eliminated if the Shuttle EVA suit system permits umbilical operations.

d. Consumables estimates include additional contingency since not all tasks require two crewmen.

e. 0.9 kg (2 lb) H_2O/hr/man + 0.03 kg (0.074 lb) O_2/hr/man = 0.93 kg (2.074 lb)/hr/man

0.93 kg (2.074 lb)/hr/man x 2 men = 1.86 kg (4.148 lb)/hr

1.86 kg (4.148 lb)/hr x 20 hr EVA = 37.2 kg (82.960 lb)

37.2 kg (82.960 lb) x 1.5 (to include contingency) = approx. 55.8 kg (125 lb) consumables

Approx. 55.8 kg (125 lb) consumables + 113.4 kg (250 lb) tankage = 169.2 kg (375 lb) total consumables (0.33 m^3 vol.)

f. Includes 20 percent allowance for packaging.

g. Includes 50 percent allowance for packaging.

astronaut enters and performs maintenance. Should EVA access to the underside of the LST be required, the manipulator will be used to lift it out of the bay, roll it 180 deg, and place it back into the bay.

Manipulator maintenance could possibly be accomplished by the Shuttle manipulator if the tray packaging concept were utilized for SSM systems. Special electrical connector(s) and guides would be required to meet the alignment requirements of the tray, and mechanical-advantage devices would have to be utilized to supplement the manipulator's limited (approximately 10 lb) force capability for engaging and disengaging the tray connectors and latches. Suited crewmen would still be required for instrument replacement (which is most likely to be the equipment requiring changeout).

Timelines for earth-return, EVA, and manipulator maintenance are provided in Chapter III.

B. Design Impacts

For earth-return maintenance, the design impacts are minimum. Easy access is the strongest design driver, and even that can be sacrificed if necessary since the LST can be partially disassembled, or access hatches can be provided in the structure if required. Ground-supplied access platforms, lights, controlled environments, and communications equipment relieve the spacecraft of the burden of supplying these for ground maintenance operations.

Three design requirements must be met to permit any degree of on-orbit EVA maintenance.

1. The suited crewmen must physically have access to the failed item.

2. Electrical and mechanical connectors must be designed to permit operation by the gloved hand or by tools the crewmen has at his disposal.

3. Handholds or footholds must be available to absorb torques and loads generated by removing and replacing the item.

In addition lighting must be provided. Assuming all conditions are met, the amount of crew time and energy required to replace a failed item will depend on the particular fasteners and connector design employed.

In the on-orbit maintenance portions of this study, it was assumed that those few items known to life-limit the LST (batteries and tape recorders) would be design-optimized for EVA maintenance. This means that electrical and mechanical fasteners would be sized for operation by the pressure-suited hand without tools and that in most cases the replacement operation would require only one hand. This design results in the least possible expenditure of crew time and energy, but includes a certain design impact to the components involved.

However, for those numerous items which are not expected to fail, but which require replacement if they do fail, a simpler design solution is assumed. This solution merely requires that fasteners and connectors be operable by an appropriate tool, and that access be provided only for the tool, not necessarily for the pressure-suited hand. The requirements on fasteners and connectors, are that all parts be captive, and that operating torques be within the (considerable) capabilities of the suited crewman. In most instances, this design philosophy requires the coordinated efforts of two crewmen to accomplish replacement of a given item. The advantage of this design philosophy is that on-orbit maintenance of a wide variety of components is made feasible with a very small design impact, at the expense of crew time and energy. The capability is thus economically provided to deal with a small number of random failures in a large group of reliable items. Design for major on-orbit EVA maintenance or for manipulator maintenance would drive the design more strongly. The main consideration is packaging of the equipment for efficient removal.

Since minor EVA maintenance has such a small impact on the LST design, it allows considerable program flexibility in selecting the degree of on-orbit maintenance at only a very small design penalty. It should be possible, for example, to extend the period before first earth-return to 5 years, with a revisit at the end of 2 1/2 years for EVA maintenance. Such an approach would necessitate designing the solar arrays, thermal control coatings, micrometeoroid protection, and optical coatings for 5 years rather than 2 1/2, which should be feasible. The list of spares and logistics in Table VI-1 should be more than ample for such a maintenance operation. Thus, elimination of the operational expenses associated with three of the earth-return maintenance visits over a 15-year lifetime should provide significant program savings.

C. Conclusions

In summary, pure earth-return maintenance results in a slightly simpler and lighter spacecraft, but has the greatest operational impacts for maintenance. This concept may permit essentially no potential for on-orbit maintenance. If, however, an earth-return maintenance concept is selected which inherently includes the access required for a suited crewman to reach the majority of the LST components, then those components may easily be made replaceable by the selection of fasteners and connectors operable by the suited crewman. The minor on-orbit maintenance capability provides considerable flexibility for the program, with minimum design impact.

CHAPTER VII. STRUCTURES

A. Structural Considerations of the SSM Open Truss Configuration

1. Introduction. Several configurations of the LST/SSM and LST/SI were generated to increase the component accessibility for EVA on-orbit maintenance. Basically, the structural systems for these configurations can be characterized by two different structural arrangements: (1) a cylindrical shell similar to the structure which was discussed in the LST Phase A documents and (2) an open truss which would also replace the main ring concept that was used in the Phase A design. Since all of the shell configurations were similar to the Phase A design with the main difference being the length and equipment mounting arrangement, this section of the report will be concerned with the new open truss only.

In this analysis it was assumed that the subsystems and SI would furnish their own meteoroid shields and of course these would not be load carrying.

2. Model Description. The analytical model of the SSM structure consists of both bar and rod members. In the analysis performed, a bar member possesses extensional, torsional and flexural stiffeners while a rod member has extensional and torsional stiffnesses only. Hence, as shown in Figure VII-1, the structure is considered to be a combination of truss (rod members) and rigid frame (bar members) members.

The main ring consists of eight bar members and forms a rigid octagonal frame. The primary LST/Shuttle interface support members are located on this main ring and are extended out to the Shuttle bay payload attachments. These members are modeled as bar members in the analysis and they are rigidly attached to the primary ring and pinned at the Shuttle attach points.

The aft frame consists of nine bar members, two of these are outrigger supports which are extended to the Shuttle attachment point.

The pinned connection between the outriggers and Shuttle attachment point reacts vertical loads (in the pitch plane) only, and this is considered as the secondary support to the LST. Therefore, the entire Shuttle attachment scheme is consistent with the standard payload/Shuttle statically determinate structural interface scheme.

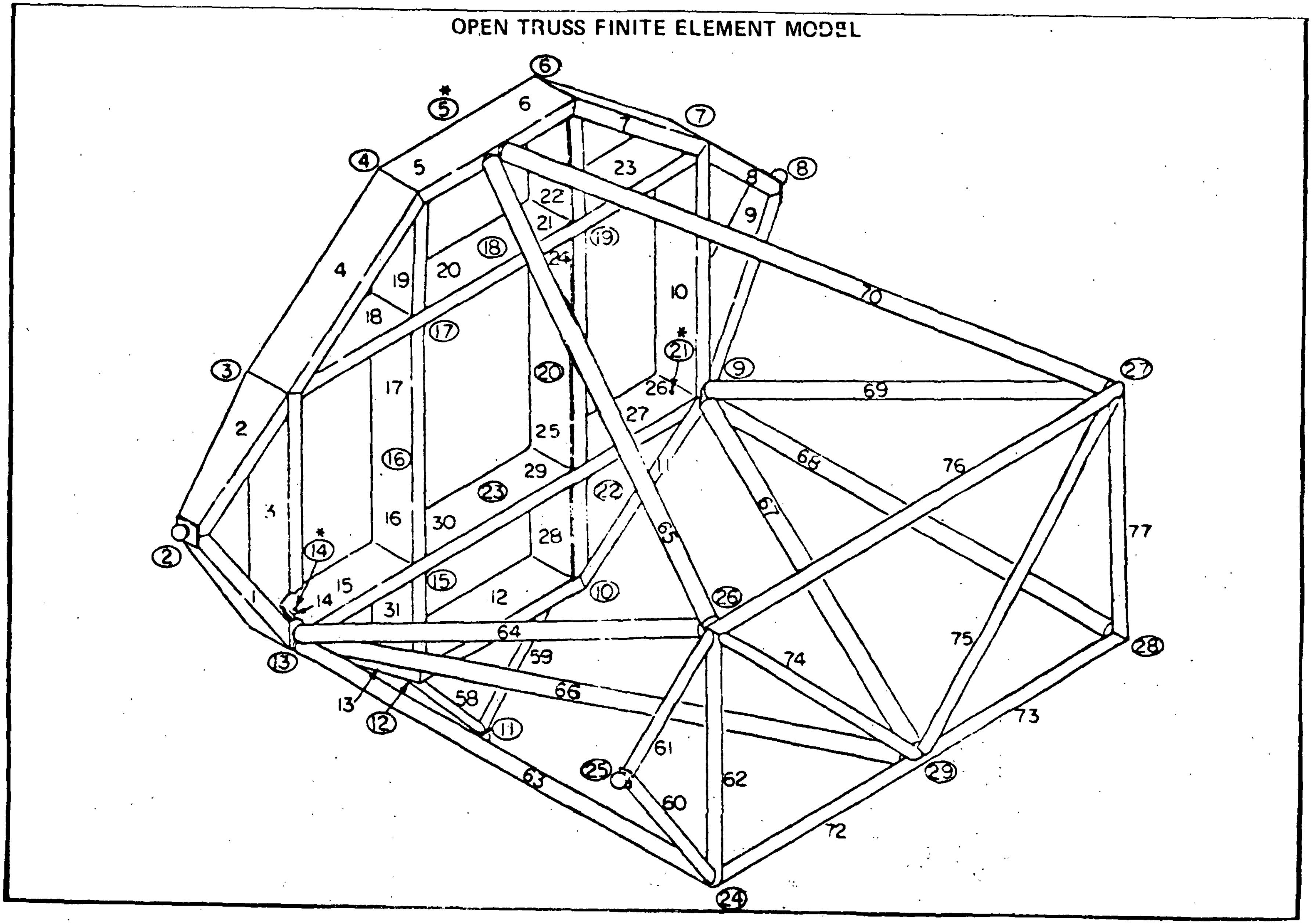


Figure VII-1. NASTRAN finite element model.

The longerons or struts between the primary ring frame and the aft frame are considered rod members which are pinned to the primary and the aft frames.

Since these longerons are relatively long members, their slenderness ratios are expected to be large and their detail design will likely be dictated by column stability requirements.

3. Design Loads. The design load conditions considered in the member designs are at booster cutoff, maximum axial acceleration, landing and crash landing. The load factors associated with these flight conditions are specified in Table VII-1. The loading conditions at booster cutoff, landing, and maximum axial acceleration are considered to be normal operations, and a factor of safety of 1.4 was applied to obtain the ultimate member design loads induced under these normal flight conditions. However, no factor of safety is applied to the crash landing load of 9.0 g in the axial direction of the LST spacecraft. At crash landing of the Shuttle, the spacecraft is not expected to survive the initial impact; however, damage to the orbiter crew compartments must be avoided by preventing large pieces of the payload from penetrating the crew compartment bulkhead. The spacecraft mass considered in the analysis is 8047 kg (17 740 lb), which includes a 10-percent contingency on all masses. An initial estimate of the SSM structural and thermal insulation weight of 504 kg (1112 lb) was also included.

TABLE VII-1. DESIGN LOAD FACTORS

Flight Phase	Load Factor (g)		
	n_x	n_y	n_z
Booster cutoff	2.50	-1.20	1.30
Maximum axial acceleration	3.25	0.45	0.85
Landing	-1.00	-0.75	-3.50
Crash landing[a]	-9.00	0	0

a. An ultimate factor of safety of 1.4 is applied except the crash landing.

4. Design Details. The 2219-T87 aluminum alloy was used for all structural members of the SSM frame shown in Figure VII-1. The primary ring frame was made of segments of 305- by 102- by 6.4-mm (12- by 4- by 0.250-in.) rectangular tubes to provide sufficient biaxial flexural stiffnesses and torsional stiffness. The interior crossbeams are made of 305- by 102- by 4.8-mm (12- by 4- by 0.188-in.) rectangular tubes, which were designed for axial force and biaxial bending moments. The design loads for all members in the primary frame are induced at crash landing.

There are three sets of LST/Shuttle attachment struts connected to the primary ring: one on each side of LST and one at the bottom. Each set consists of two struts which connect to a support point in the Orbiter cargo bay. The struts on each side of the payload are designed to transmit the longitudinal and vertical loads to the Shuttle and are made of segments of 305- by 102- by 8-mm (12- by 4- by 0.312-in.) rectangular tubes, which were sized for combined axial force, biaxial bending, and torsional moments. The struts at the bottom of the primary ring and at the side of the aft frame are made of circular tubes of 64-mm (2.5-in.) OD by 2-mm (0.078-in.) wall. These two sets of struts are sized for landing loads.

The longerons between the main and the aft frames are all made of circular tubes of 102-mm (4-in.) OD by 4-mm (0.156-in.) wall. All of these members are pin connected to the frames, and are designed for column stability. The design loads for most longerons are induced at crash landing. However, the normal landing and booster cutoff conditions also dictate the designs of a few longeron members.

5. Main Frame Deflection. The eccentricity of 356 mm (14.0 in.) between the resultant reaction of the longitudinal loads and the longitudinal axis of the LST causes some concern for the main frame deflections in the longitudinal direction. The deflection manifests itself in two forms: (1) the rigid-body rotation of the frame about a pitch axis connecting the two attach points and (2) the warping and bending of the crossbeams.

Figure VII-2 shows the total deflection and warping of the primary ring for the maximum axial acceleration and crash landing conditions. The solid line connects the total longitudinal deflections of each node of the ring. The dotted lines represent the rotation of the frame. Therefore, the difference between the solid and dotted lines is the amount of warping of the ring. The maximum warping is approximately 13 mm (0.5 in.) at orbiter engine cutoff and 36 mm (1.4 in.) at crash landing with the 9-g axial acceleration.

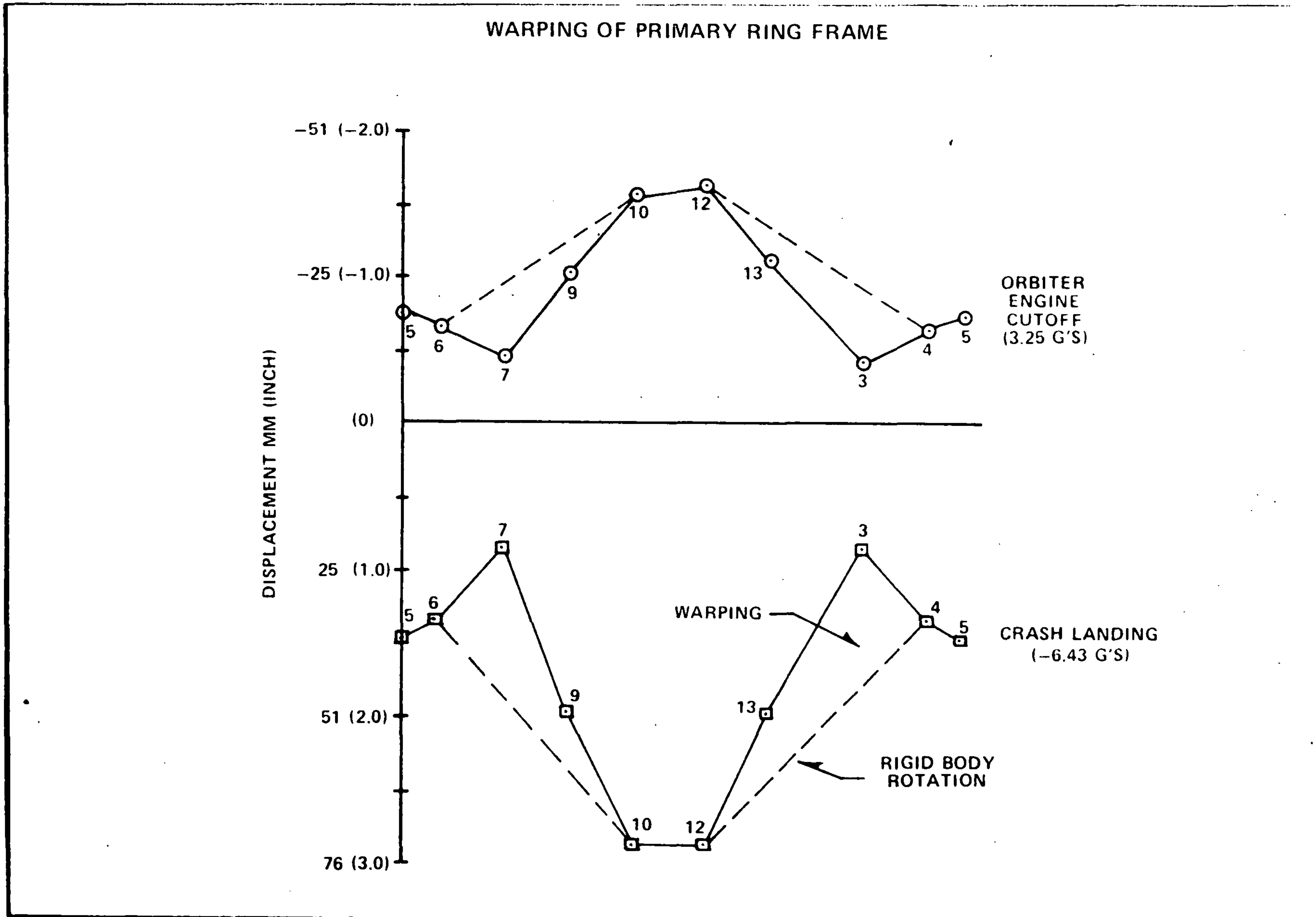


Figure VII-2. Warping of primary ring frame.

The bending of the primary frame is represented by the deflections of the crossbeams, as shown in Figure VII-3. Since the frame is supported on both sides in a yaw plane, the horizontal crossbeams bend more than the vertical crossbeams. The deflection of the vertical beam, which is almost straight lines, indicates that the displacements are primarily rigid-body rotations of the primary frame about the pitch axis through the two forward tie points. The primary cause of the rigid-body rotation is the eccentricity between the payload longitudinal axis and the resultant support reactions.

6. Weight Summary. A summary of the basic structural weight is shown in Table VII-2. The weights shown for the major members are based on the results of the static structural analysis while the end fitting and equipment support fitting weights are estimated.

B. Thermal Distortion Analysis

1. Introduction. Since the open truss design is a totally different concept thermally, a thermal distortion analysis was performed to obtain not only the structural displacements but the forces acting on the mirror through the mirror mounts. These forces were then compared with the resulting forces from the Phase A design.

2. Structural Model. The NASTRAN finite element structural model (Fig. VII-1) used for the static analysis was also used for the thermal distortion analysis. However, some refinements were made in the modeling of the primary mirror mounting structure. Figure VII-4 depicts the primary mirror flexible mounts that were added to the model. The actual modeling of the flexible mirror support mounts was accomplished in the NASTRAN program by connecting the main frame grid points 5, 14, and 21 to the flexible mount grid points 1, 30, and 31, respectively, through the use of multipoint constraint equations (MPC). The mount grid points 1, 30, and 31 were connected to the primary mirror and flexible mount grid points 32, 33, and 34 by the Phase A Invar flexures [254-mm (10.0-in.) by 344-mm (13.56-in.) by 7-mm (0.27 in.) thick rectangular bars].

3. Temperature Distribution. The thermal gradients on the structure were obtained from the thermal analysis discussed in this report.

Since adequate computer time was not available to complete the temperature history, the computation was stopped before the temperature stabilization was achieved. Stabilized values were obtained, however, by extrapolating the nonstabilized values.

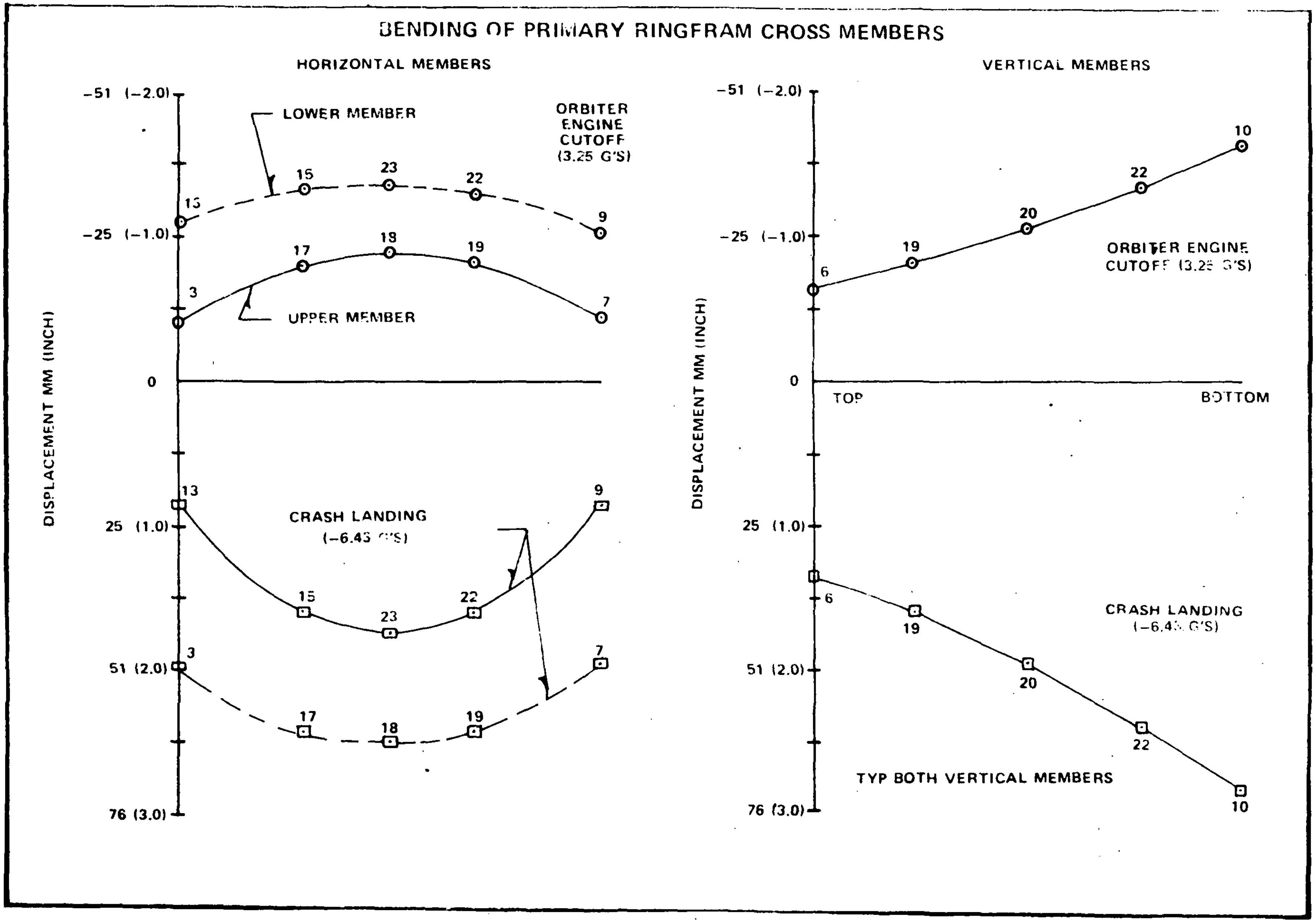


Figure VII-3. Bending of primary ring frame cross members.

TABLE VII-2. STRUCTURAL WEIGHT SUMMARY

Item	Wt. kg	Wt. lb	Totals kg	Totals lb
Primary Frame			349	769
Ring segment	147	323		
Cross stiffeners	132	291		
Fittings and splices	70	155		
Aft Frame Assy			54	119
Aft frame struts	40	89		
Fittings	14	30		
LST/Shuttle Attach			112	246
Interface struts	84	184		
Attach fittings	28	62		
Longerons			111	245
Struts	95	210		
Fittings	16	35		
Equip. Attach			90.7	200
Fittings and structural strengthening	91	200		
Total			716	1579

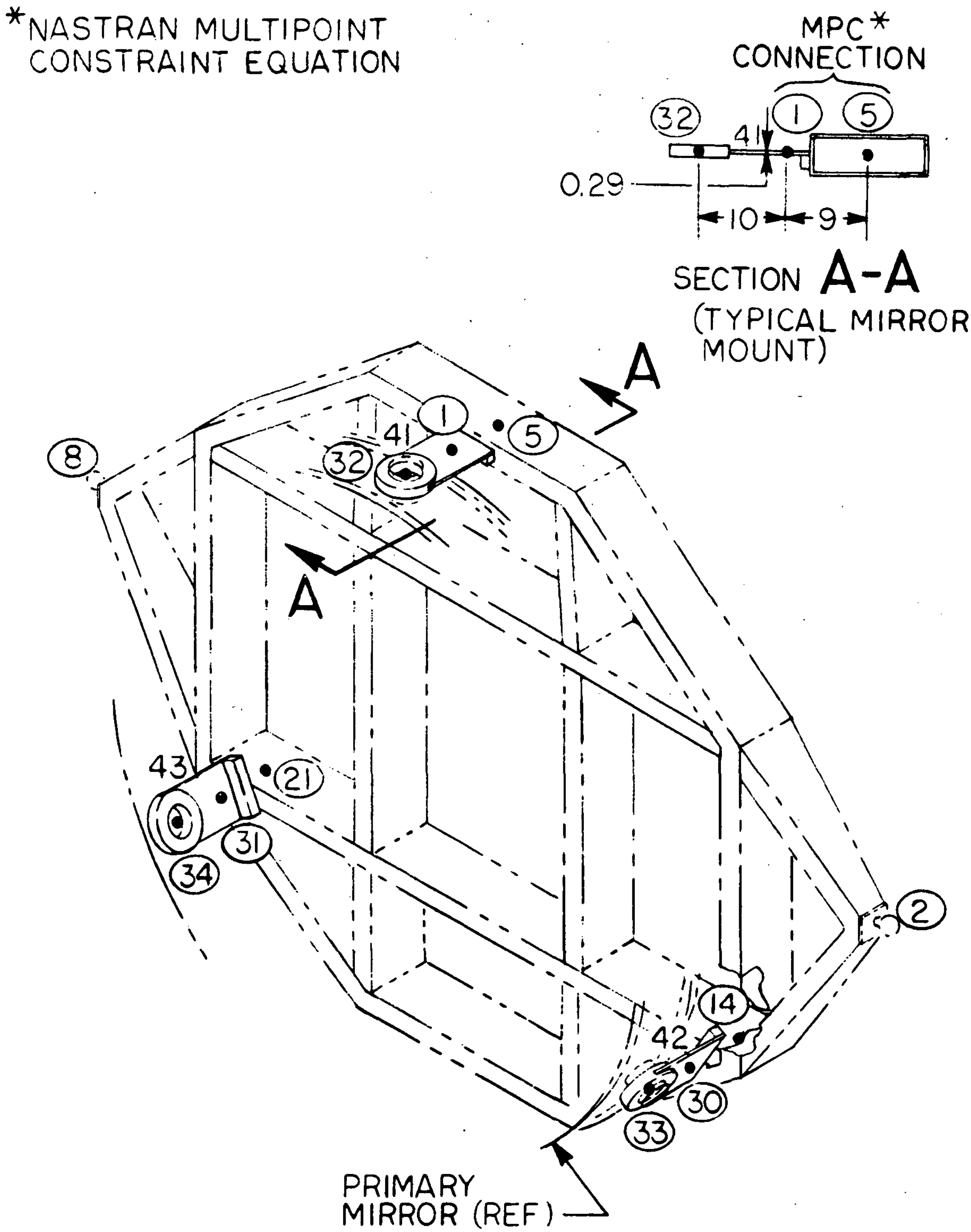


Figure VII-4. Finite element model of primary mirror flexible mounts.

Since the nonstabilized values represent a realistic case, the thermal distortion analysis includes both the stabilized and nonstabilized cases and Table VII-3 gives the stabilized and nonstabilized temperatures for each member in the analysis.

4. Mirror Loads. As mentioned previously, a prime objective of the thermal distortion analysis was to determine the resultant loads induced into the primary mirror by the thermal gradients.

Since the mirror has high in-plane stiffness, its in-plane distortion is negligible. With this fact and the existence of the determinate mount concept, it is possible to represent the mirror by three simple rigid supports and determine the resultant load reactions.

The computed radial and tangential forces at the mirror supports are given in Table VII-4. The axial forces at the supports are negligible.

The approximate maximums of these radial forces are compared in Figure VII-5 with the forces obtained for the Phase A design. As can be seen, the stabilized temperatures yield forces that are much too large.

5. Thermal Distortion. Table VII-3 summarizes the translational components of the displacements caused by the thermal gradients.

C. Conclusions

As expected, the static analysis yielded loads, deflections, and stresses that were reasonable; however, the static analysis seldom points out the problems associated with the truss type structures used to fulfill the stringent requirements of LST and similar designs. These are typically adequate fitting design and equipment support design.

The thermal distortion and induced loads were greater than anticipated. However, the magnitudes of these parameters are directly related to the thermal gradients and materials used in the design. Little can be done with the basic design to lower the values or improve the performance with the exception of employing a link-type mirror support that would not transmit radial loads to the mirror.

Many things can be done with the thermal protection and materials to lower these values. It is obvious that an adequate design can be generated to

TABLE VII-3. MEMBER TEMPERATURE SUMMARY

NASTRAN Member Number[a]	Stabilized Temperature		Nonstabilized Temperature	
	°C	(°F)	°C	(°F)
1	-19.4	(-3.0)	5.6	(42.0)
2	-19.4	(-3.0)	5.6	(42.0)
3	-18.3	(-0.9)	6.7	(44.1)
4	-18.9	(-2.0)	6.1	(43.0)
5	-18.3	(-1.0)	6.7	(44.0)
6	-18.3	(-1.0)	6.7	(44.0)
7	-16.1	(3.1)	8.9	(48.1)
8	-16.9	(1.5)	8.1	(46.5)
9	-14.5	(5.9)	10.5	(50.9)
10	-16.3	(2.7)	8.7	(47.7)
11	-15.9	(3.4)	9.1	(48.4)
12	-17.3	(0.8)	7.7	(45.8)
13	-18.6	(-1.4)	6.4	(43.6)
14	-18.1	(-0.5)	6.9	(44.5)
15	-18.1	(-0.5)	6.9	(44.5)
16	-18.6	(-1.4)	6.4	(43.6)
17	-18.8	(-1.8)	6.2	(43.2)
18	-18.8	(-1.9)	6.2	(43.1)
19	-18.8	(-1.8)	6.2	(43.2)
20	-18.8	(-1.9)	6.2	(43.1)
21	-16.2	(2.8)	8.8	(47.8)
22	-17.9	(-0.2)	7.1	(44.8)
23	-16.2	(-2.8)	8.8	(47.8)
24	-17.9	(-0.2)	7.1	(44.8)
25	-17.3	(0.9)	7.7	(45.9)
26	-16.3	(2.7)	8.7	(47.7)
27	-16.3	(-2.7)	8.7	(47.7)
28	-17.3	(0.9)	7.7	(45.9)
29	-16.3	(2.7)	8.7	(47.7)
30	-18.1	(-0.5)	6.9	(44.5)
31	-18.6	(-1.4)	6.4	(43.6)
53	-17.8	(-0.1)	7.2	(44.9)
59	-16.4	(2.4)	8.6	(47.4)
60	-16.3	(2.7)	8.7	(47.7)
61	-16.4	(2.4)	8.6	(47.4)
62	-15.4	(4.2)	9.6	(49.2)

TABLE VII-3. (Concluded)

NASTRAN Member Number[a]	Stabilized Temperature		Nonstabilized Temperature	
	°C	(°F)	°C	(°F)
63	-17.5	(0.5)	7.5	(45.5)
64	-16.7	(1.9)	8.3	(46.9)
65	-15.0	(5.0)	10.0	(50.0)
66	-15.7	(3.8)	9.3	(48.8)
67	-13.7	(7.3)	11.3	(52.3)
68	-14.1	(6.6)	10.9	(51.6)
69	-11.9	(10.6)	13.1	(55.6)
70	-11.4	(11.4)	13.6	(56.4)
72	-13.4	(7.8)	11.6	(52.8)
73	-13.4	(7.8)	11.6	(52.8)
74	-13.9	(6.9)	11.1	(51.9)
75	-12.7	(9.1)	12.3	(54.1)
76	-13.0	(8.6)	12.0	(53.6)
77	-11.0	(12.2)	14.0	(57.2)

a. See Figure VII-1.

satisfy the structural requirements, but it will most likely be just adequate resulting in very little margin and, in operation, it will demand very close adherence to the nominal design values. Furthermore, the design will probably require thermal shields that will alleviate some, if not most, of the strived for direct accessibility.

TABLE 4. PRIMARY MIRROR LOADS CAUSED BY THERMAL GRADIENT

NASTRAN Joint Number[a]	Stabilized Temperatures				Nonstabilized Temperatures			
	Radial Force[b]		Tangential Force[c]		Radial Force[b]		Tangential Force[c]	
	newton	(lb)	newton	(lb)	newton	(lb)	newton	(lb)
32	339	(76.17)	10	(2.19)	125	(28.08)	10	(2.19)
33	341	(76.63)	-9	(-1.96)	123	(28.64)	-11	(-2.45)
34	324	(72.83)	-1	(-0.23)	106	(23.85)	1	(0.27)

a. See Figure VII-2.

b. Radial force positive inward.

c. Tangential force positive counterclockwise as viewed from main frame to primary mirror.

CHAPTER VIII. THERMAL CONTROL

A. Introduction

Two LST concepts were investigated, (1) an open truss concept, and (2) a concept utilizing a shell-type structure similar to the Phase A design. The open truss concept required the greatest amount of analysis, since it was the most different from the previous LST concepts. A brief investigation of the applicability of heat pipes was also performed and is documented in this section.

B. Discussion

1. Open Truss LST Concept. Because of the mission time involved it is desirable to keep the thermal control technique as passive as possible to increase reliability. Certain thermal control techniques are particularly suited to equipment with wide temperature requirements. More complex techniques are needed for the narrower requirements.

For the open truss structure concept, a combination of insulation and surface coatings was used. This design is inefficient power-wise, because heaters are required. The system must be designed to dissipate the heat load in the hottest orbit and orientation. While in a cold orbit, heaters are required to maintain a minimum allowable temperature. Louvers or variable conductance heat pipes can be utilized to reduce the heater power requirement.

2. Open Truss SI Evaluation. The basic model used to analyze this configuration is presented in Figure VIII-1. Each longitudinal truss member and forward cross member was broken into two nodes. The remaining truss members were modeled as one node each. Each instrument was modeled as one node. Each cylindrical cowling over the light path from each instrument was modeled as one node. The cowling from the spectrograph select assembly to the backing for the primary mirror was broken into 12 nodes.

In the Phase A design, the pressure shell and meteoroid shield acted as a thermal buffer between the instruments and the external environment. This buffer is desirable to prevent orbital excursions. Therefore, in the open truss design, it must be installed around the individual instruments and light paths. In the thermal model, a selected thickness of insulation was incorporated around each instrument and truss member. It is required

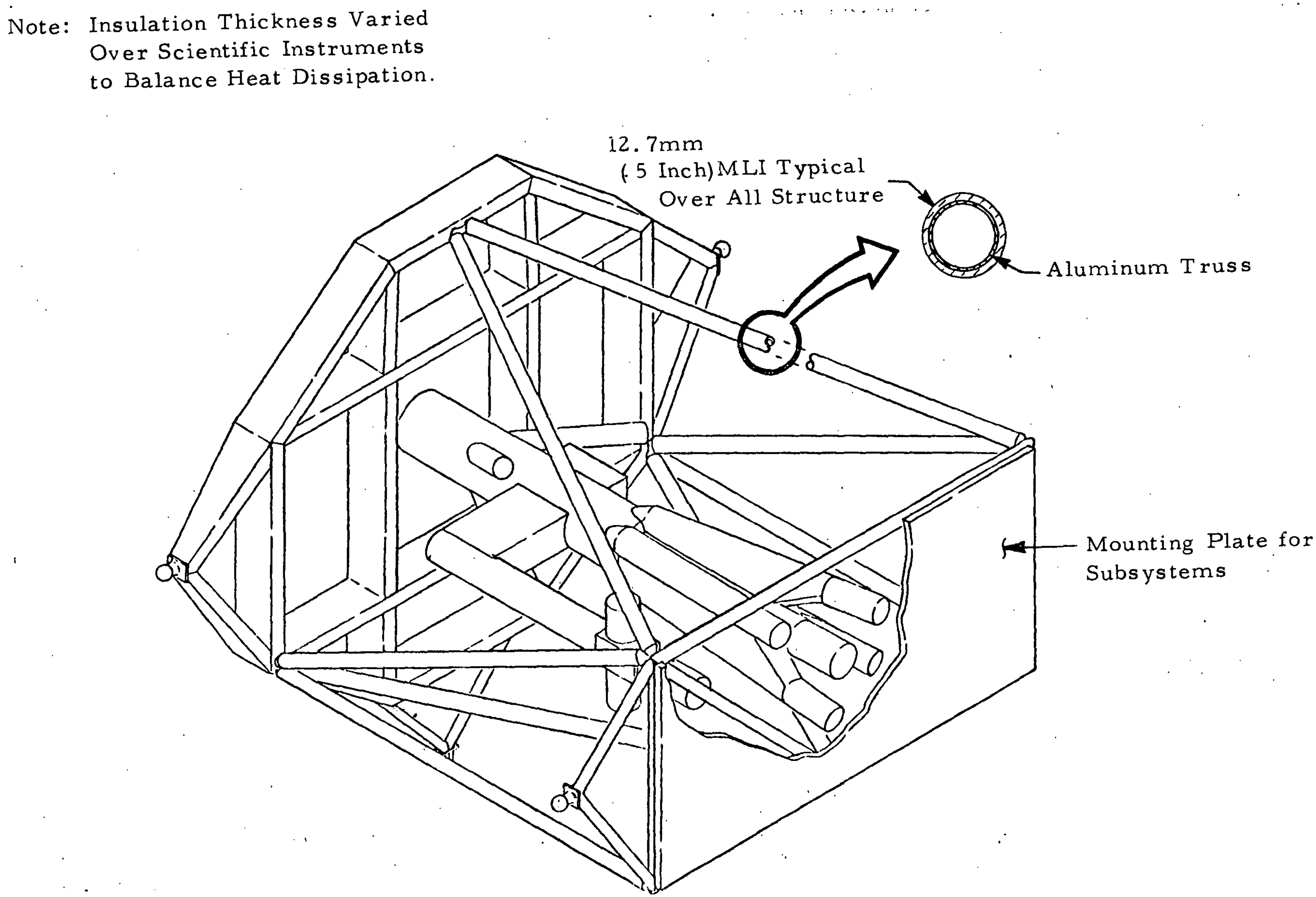


Figure VIII-1. Open truss structural model utilized for analysis.

around the truss members to prevent structural deformation caused by temperature variations and differences from side to side of the structure. For the truss members, 12.7 mm (0.5 in.) of multilayer insulation was used. The thermal conductivity of urethane foam, K = 1.72 J/m·sec·K

(K = 0.0277 $\frac{\text{BTU}}{\text{hr-ft-}^\circ\text{F}}$) was assumed for insulation of the scientific instruments. Thickness was varied based on surface area and heat dissipation rate. Insulation thickness used is as follows:

Instrument Enclosed	Thickness	
	mm	(in)
F-96 Camera	2.0	(0.08)
Cowling Enclosing F-12 Camera	3.8	(0.15)
FOS Camera	3.3	(0.13)
Echelle	3.3	(0.13)
Cowling Enclosing, Fine Guidance Figure Sensor Focus Sensor	13	(0.5)

The absorptivity and emissivity values used on the external surfaces are as follows:

Nodes	α	ϵ
All trusses	0.33	0.75
F-96 Cameras	0.135	0.92
F-12 Cameras	0.135	0.92
FOS	0.35	0.85
Echelle	0.35	0.85
F-96 Camera Arm	0.35	0.85
Cyl. over figure sensor and Fine Guidance	0.35	0.85

Heat that was assumed to be dissipated and the temperature requirements are presented in Table VIII-1.

TABLE VIII-1. HEAT DISSIPATED AND TEMPERATURE REQUIREMENTS OF SCIENTIFIC INSTRUMENTS

S. I.	No. of Units	Unit Heat Dissipation (W)	Temperature Requirement	
			Orb. Variation (°C)	Maximum (°C)
F-96 Camera	3	50	±4	50
FOS Camera	4	22	±4	50
Echelle	2	22	±4	50
F-12 Camera	1	50	±4	50
Fine Guidance	1	30	±4	50
Figure Sensor	1	6	±4	50
Focus Sensor	1	6	±4	50

It is desirable for the cameras to operate at low temperatures; therefore, an attempt was made in this model to lower the temperature of the instruments and still prevent excessive orbital excursions. This was done by using the thin layer of insulation with a conductivity such as that of urethane foam and an external coating with low solar absorptivity and high infrared emissivity characteristics. The approach is to utilize the high emissivity value over the camera surface area to overcome the heat load and lower the temperature to a desirable operating range, while the heat capacity of the cameras plus the small influence from solar radiation (because of low α) dampens the orbital cycling. Although the infrared absorptivity is high, there is little variation in the heat absorbed in an orbit. This is because with the open truss configuration the instruments can view the Earth at almost any position in orbit.

The model was put into a simulated 611-km (330-n. mi.) circular orbit with the beta angle set at 52 deg. Vehicle orientation was with the longitudinal axis perpendicular to the solar vector.

The Chrysler Shape Factor Program was utilized to calculate the geometric shape factors between surfaces. Environmental heating rates of the external surface were predicted using the Lockheed Orbital Heat Rate Program. The thermal response of the spacecraft to the environmental heating rates and the internal heat sources were evaluated through utilization of the SINDA digital computer program.

No significant temperature fluctuations occurred in the truss structure members because of orbital excursions. There were gradients along the length of some of the members and also gradients from side to side of the entire truss. Truss members extending fore and aft showed gradients from end to end of 1.1 °C (2° F) to 2.8 °C (5.7° F). Maximum gradient from the primary ring to aft members was 5.6 °C (10° F). From solar side members to antisolar side members the maximum gradient was - 6.7° C (12° F). Although the program was not run long enough for the temperatures to stabilize, extrapolation indicated they would settle at approximately -17.8° C (0° F). With the insulation thickness and thermal coating characteristics used in this analysis, the instruments ran at temperatures in the vicinity of -17.8° C (0°F). The only significant orbital fluctuation occurred in the FOS cameras. The maximum was 4.5° C (8° F).

3. Open Truss Support Systems Module Evaluation. In the open truss configuration, the vehicle subsystems are located in modules on a platform at the aft end of the SI structure. Except for the batteries, all the subsystem components have a broad operating temperature tolerance. A preliminary analysis was performed on two modules attached to the platform in a 611-km (330-n. mi.) circular orbit with the beta angle set at 52 deg. This investigation was performed with insulation around each individual module, and with a shroud over the modules as shown in Figure VIII-2. Two cases were investigated for each configuration, one with the platform perpendicular to the solar vector and another with the platform parallel to the solar vector. The external surfaces (individual module and external shroud) were assumed to be coated with zinc-orthotitinate, which has an absorptivity and emissivity value of 0.135 and 0.92, respectively. For the shroud cover cases, the emissivity for both the modules and the shroud internal surface was assumed to be 0.9. The two modules selected were: (a) 635 mm (25 in.) by 635 mm (25 in.), containing three regulators and one electrical control assembly, and (b) 381 mm (15 in.) by 508 mm (20 in.), containing one battery and one charger. Modules (a) and (b) have a heat dissipation rate of 75 W and 27 W, respectively. In the calculations for the cases which include a shroud, the modules' surfaces radiated directly to the shroud. It was assumed the shroud was an aluminum sheet with no insulation, but coated to match the above thermal characteristics. For the cases without the shroud, a layer of urethane foam was assumed to be over each module. The thicknesses of the layer were 1.5 mm (0.058 in.) and 2.4 mm (0.096 in.) for modules (a) and (b), respectively.

Resulting module temperatures were as presented in Table VIII-2.

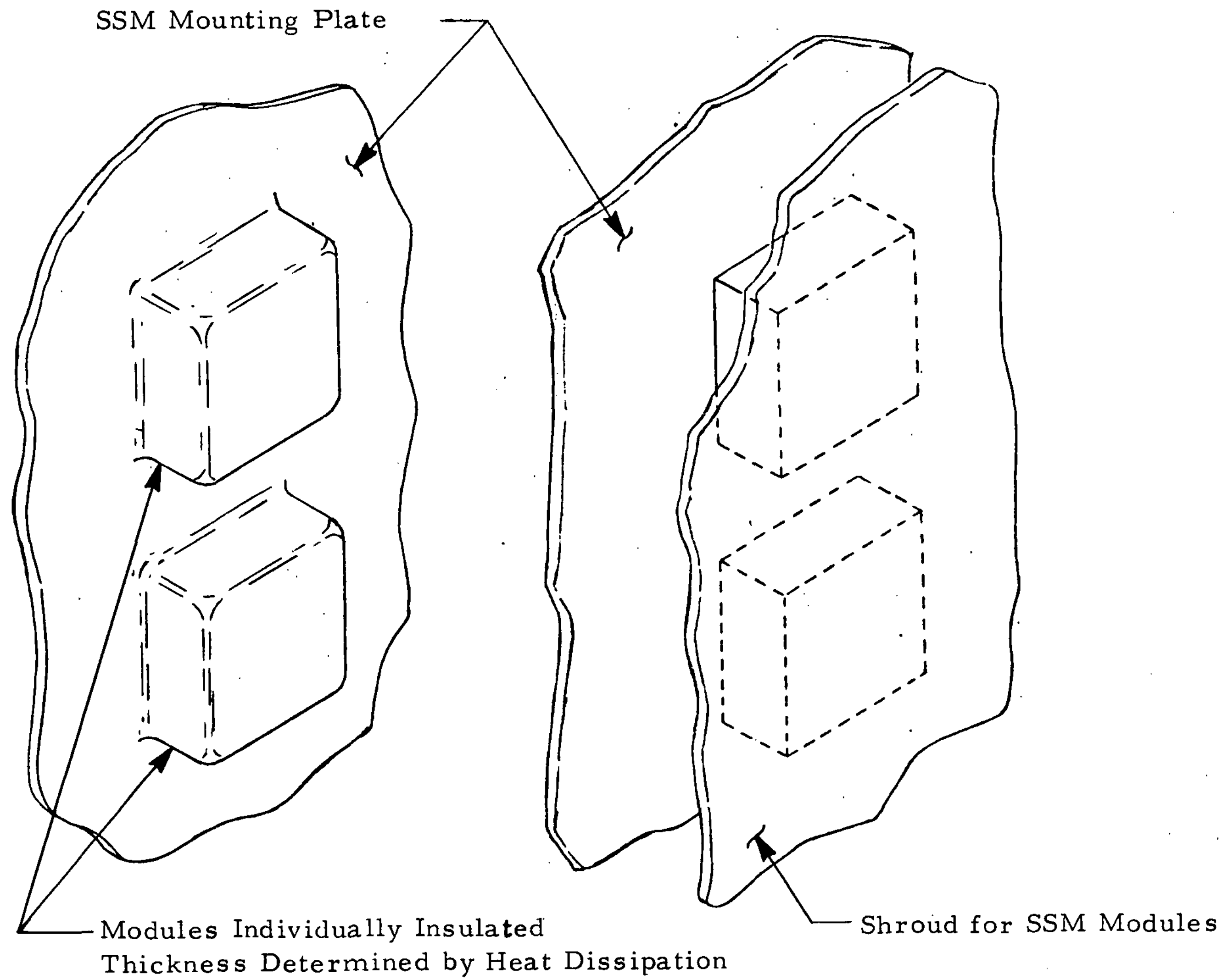


Figure VIII-2. Thermal model utilized for preliminary analysis of typical subsystem modules located on aft mounting plate.

TABLE VIII-2. TYPICAL SSM MODULE TEMPERATURES

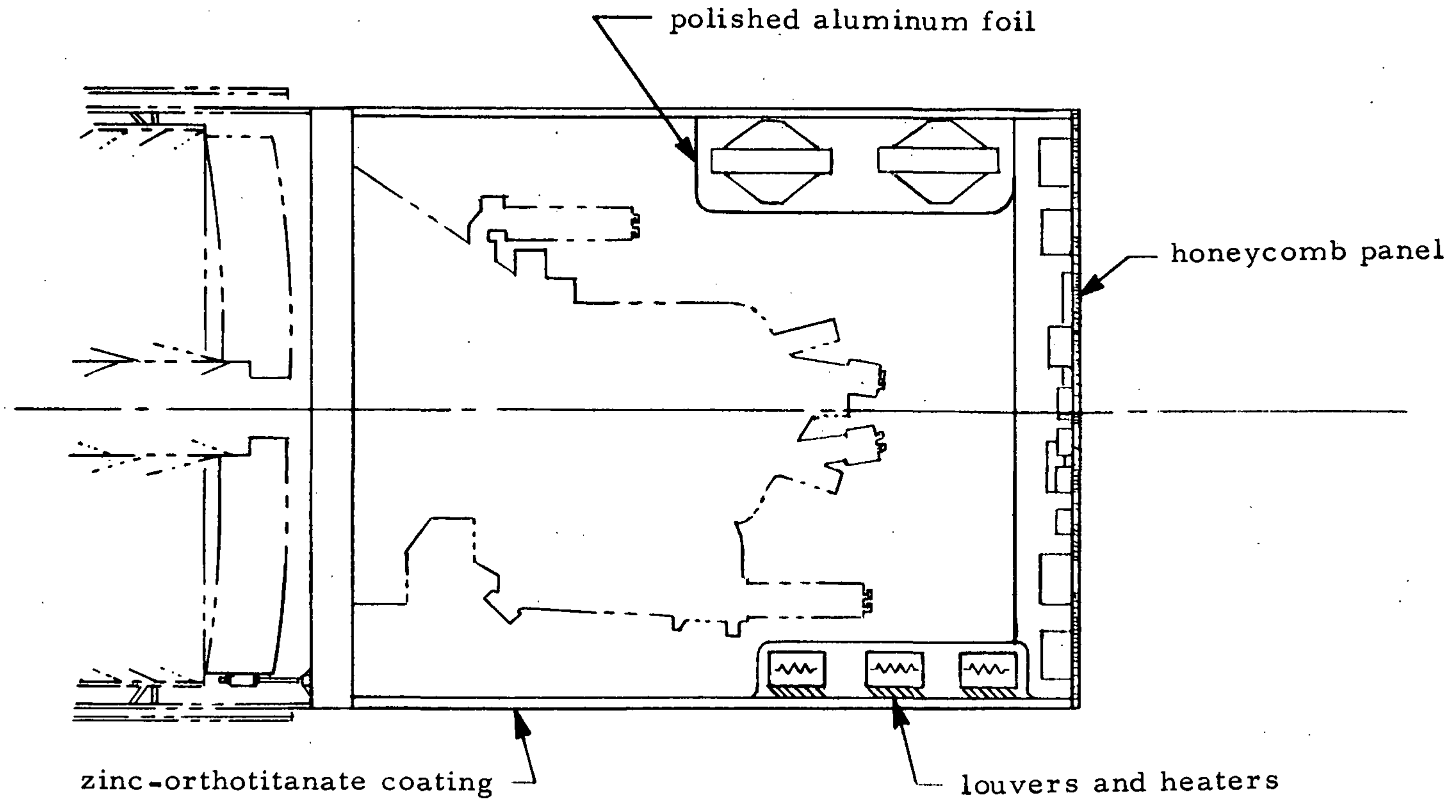
Module		Temperature, °C (°F)		
		Beta = 52 Perp.	Beta = 52 Par.	ΔT
a	Without Shroud	29.4 (85)	0.56 (33)	28.8 (52)
b		22.2 (72)	-6.7 (20)	28.9 (52)
a	With Shroud	36.1 (97)	15 (59)	21.1 (38)
b		26.7 (80)	3.3 (38)	23.4 (42)

With the shroud incorporated, the resulting temperature difference caused by orbit plane orientation change is 7.7°C (14°F) to 5.5°C (10°F) less for modules (a) and (b), respectively.

C. Shell LST Concept

The shell concept is very similar to the Phase A LST design. From the thermal control standpoint, the principal difference is in the location of the SSM components, which are located on the inside of an aft honeycomb panel and directly inside the shell. The SI components are located on the SI truss, as shown in Figure VIII-3. An analysis has not been performed on this particular configuration but analyses on similar configurations show that thermal control can be accomplished through the use of components and features similar to those utilized in the Phase A design:

- Coatings
- Radiator plates
- Louvers
- Insulation
- Polished aluminum foil
- Thermostatically controlled heaters
- Component grouping
- Isolated battery compartments



Note: Heaters typical on all components.

Figure VIII-3. LST SI and SSM configuration showing thermal control concept.

A thermal barrier must be maintained between the SI and SSM components. This can be accomplished by an insulating blanket, as in the Phase A design. If the blanket induces excessive contamination, an alternative is the use of a metallic foil with a very low surface emissivity.

Temperature control is accomplished by establishing a heat balance between the absorbed radiation (solar, albedo, and earth), internal heat dissipation, and emitted energy. This balance must be designed for the hottest orbit and vehicle orientation. Heaters and/or louvers are necessary to prevent the components from dropping below the lower temperature limit while in the cold orbit and vehicle orientations for extended periods of time. The heat capacity of the system, the insulation, and the coatings tend to minimize the effect of large variations in incident radiation caused by changes in orbital parameters or vehicle orientation.

All SSM components except the batteries have a broad operating temperature limit. With these broad limits very little heater power is required. But the batteries, which experience large variations in internal heat dissipation and also have a +5 to +15°C temperature requirement, must be thermally controlled closely with heaters and louvers. When the batteries are not dissipating the heat load for which the thermal control system was designed, an equivalent amount of heat must be input by heaters to hold the temperature within limits. Louvers between the battery and the external environment can alleviate the requirement for a large amount of this heater power.

It is desirable for the cameras to operate at low temperatures; therefore, the design for the SI system should balance the heat dissipation and emission so that the cameras will be at as low a temperature as possible without reflecting temperature changes caused by orbital excursions. This can be accomplished by utilizing zinc-orthotitanate coating on the external SSM surface and a coating with high emissive characteristics on the component and on the inside SSM wall.

To lower the temperature of the components the shell wall temperature must be lowered. Analyses have not been performed to determine the temperatures that can be expected, but it is obvious that with any stiff structural attachment the cold shell will put structural and thermal stresses into the telescope primary mirror. A flexible or hinged attachment between the shell and the mirror incorporating an insulating block would decrease the seriousness of this problem.

The shell approach was selected as the recommended one for the Phase A update design since it provides better and easier environmental protection for the LST equipment with less heater power required and since it is a less critical design, it may reduce the amount of testing required.

D. Heat Pipe Comparisons

The present thermal control design, which utilizes limited heaters and louvers in addition to the completely passive elements, is adequate for the current LST requirements and provides considerable margin for growth. Also, such a passive design provides high reliability for the system. However, extensive design changes could render such a system more costly in the long run than a more active system since its limited dynamic range would require more thermal-vacuum testing than a system with a greater dynamic range.

For these reasons, plus the fact that it is desired to operate the image tubes at even colder temperatures than the Phase A design, a brief investigation of heat pipes in comparison with other designs was undertaken.

Heat dissipated by the subsystem equipment must be transferred to the external surface of the spacecraft by radiation or conduction (direct or heat pipe). Under standby or minimum load conditions, a method is required to keep equipment from becoming too cold. A passive system will be supplemented by heaters. Louvers will close to block the radiation path and should be used if the heater power becomes large or if a narrow temperature range is required. The current design uses louvers for thermal control of the batteries and passive radiation with heaters for the remaining subsystem equipment.

An example of controlling avionic equipment to a temperature range of 35° to 85° C is shown in Figure VIII-4. The figure shows that a power range of 158/95.8 W/m^2 (14.6/8.9 W/ft^2) is required by a passive system, about a 3-to-1 variation can be handled by louvers, and close to 11-to-1 variation can be handled by a variable conduction heat pipe (VCHP). Because of this wide range the VCHP is flexible to changes in heat load, to location of equipment, and to potential growth as well as being insensitive to environmental changes. It would also reduce the amount of thermal vacuum testing required since only the boundary conditions would have to be examined. This could be a large cost saving to the LST program. By standardizing modules, a cost saving in thermal control subystems employing VCHP could be realized. This saving is shown in Table VIII-3. This should be investigated further in Phase B.

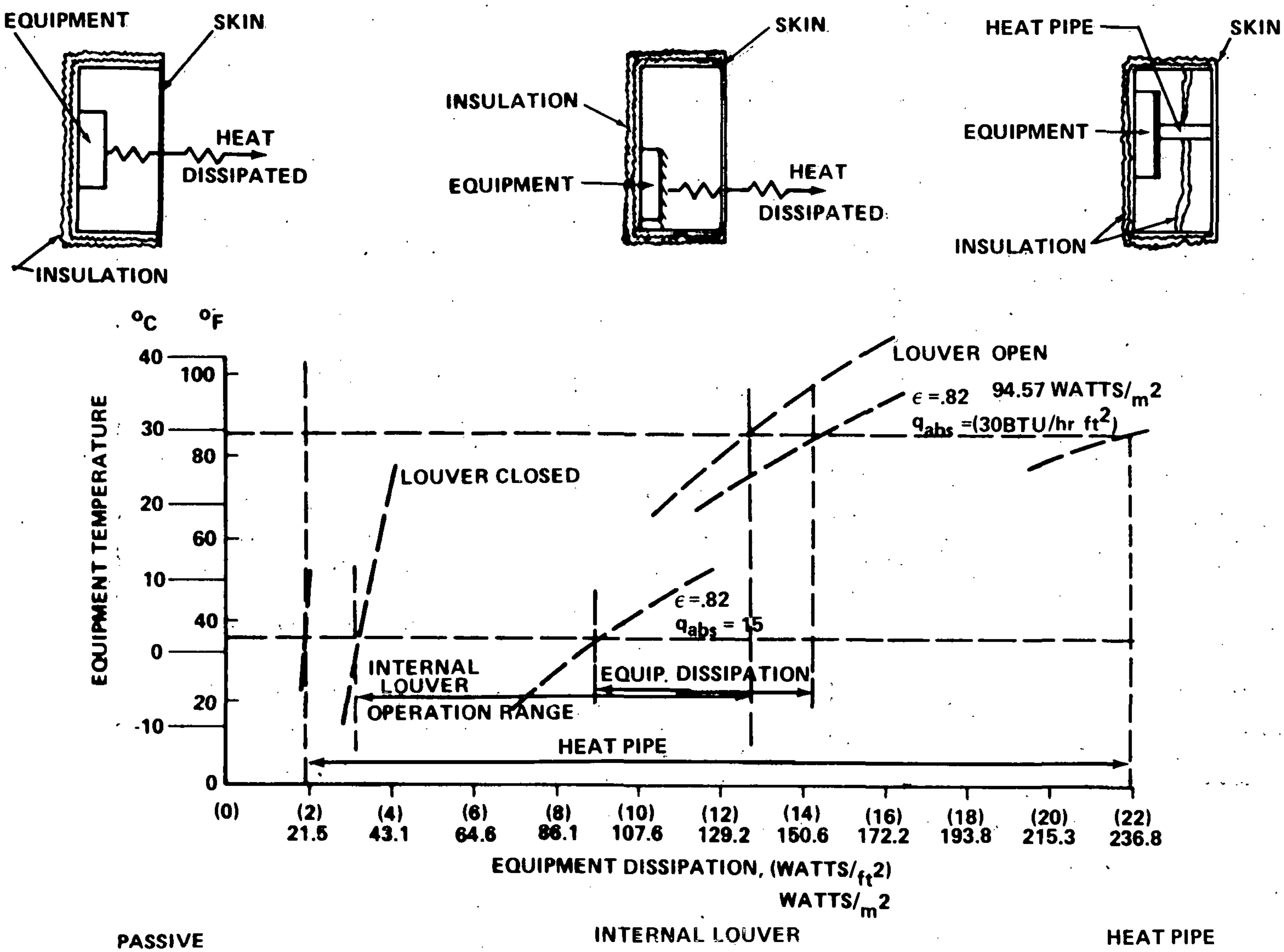


Figure VIII-4. Thermal control range comparison.

TABLE VIII-3. THERMAL CONTROL COST COMPARISON

	Passive($)	Louver($)	VCHP($)
First unit cost	600.	13 500.	2 650.
95-percent learning curve			
100 units cumulative	45 952.	1 110 000.	203 100.
100th unit	427.	9 500.	1 880.
Average price	460.	10 100.	2 031.
90-percent learning curve			
100 units cumulative	34 885.	770 000.	153 000.
100th unit	298.	6 600.	1 320.
Average price	349.	7 700.	1 530.
85-percent learning curve			
100 units cumulative	26 252.	585 000.	116 100.
100th unit	204.	4 500.	900.
Average unit	263.	5 850.	1 161.

E. Conclusions

With the open truss structure hard-mounted to the primary mirror and the structure dropping to -17.7° C (0° F), excessive stress is encountered. For this preliminary analysis absorptivity and emissivity values chosen for the coating over the insulation were 0.33 and 0.75, respectively. This $\frac{\alpha}{\epsilon}$ ratio is 0.44. It is obvious that a coating must be selected with a higher $\frac{\alpha}{\epsilon}$ ratio to increase the truss structure temperature and thereby decrease the stress on the mirror. Several analysis runs would be required to zero in on the proper ratio. It is noted that most coatings degrade with time and allowances must be made for long-term missions. It is concluded that the scientific instruments can be maintained at a low temperature (17.7° C) and not fluctuate beyond the allowable limit. This can be done with the zinc-orthotitanate coating and a balance of insulation and heat dissipation. Design calculations must be made for the hottest case and heaters must be utilized to maintain temperature for the colder cases. Although this analysis indicates that the open truss design is feasible, it is noted that thermal design is more critical than with the shell configuration; therefore, more testing would be required.

With the exception of the batteries, all subsystem equipment can be thermally controlled with or without a shroud by selection of coatings and heaters. For most components, very little heater power will be required because the minimum temperature limit is low. If an operating mode requires the SSM mounting plate to face the sun, more surface area will be required for the battery modules to keep the temperature below the maximum limit. If these radiating surfaces never face the sun, control for maximum temperature limits can be accomplished with the presently designed modules. For cold orbits and cold vehicle orientations, heaters and louvers are recommended to keep the battery temperature above the minimum limit. The heaters are a necessity, and the louvers are recommended to conserve power. Table VIII-4 is a summary of advantages and disadvantages of the truss- and shell-type configurations.

The utilization of heat pipes as a potential cost-saving approach, particularly in the area of thermal-vacuum testing, should be investigated further.

TABLE VIII-4. TRUSS VERSUS SHELL THERMAL CONTROL

Configuration	Advantages	Disadvantages
Truss and/or open configuration (passive)	Space saving Ease of mounting and maintenance	Sensitive to changes in: Heat loads Location Vehicle orientation and orbit geometry Small heat capacities Maximum thermal vacuum testing Growth is limited Heaters required
Shell with micrometeoroid (thermal) shield (passive)	Flexible to limited changes in: Heat loads Location Growth Less sensitive to vehicle orientation and flight geometry Large heat capacity Less heater power req'd	Matching of electronic equipment with thermal loads Thermal vacuum testing Limited heaters required

CHAPTER IX. ELECTRICAL SYSTEMS

A. Guidelines and Requirements

The electrical power system defined herein is a revised version of the system defined in the Phase A report. The guidelines applicable to this system are identical to those defined in the Phase A analysis except for those listed below:

1. Earth return for maintenance will be the primary mode for refurbishment and/or repair.

2. The maximum period elapsing before the initial refurbishment operation will be 2 1/2 years.

3. Launch and retrieval will be by way of the Shuttle vehicle only.

4. Off-sun pointing for a maximum of three consecutive orbits to occur no more than once per day is a design goal.

In addition to these new requirements, the system was analyzed to determine if complexity and cost could be reduced and maintain sufficient reliability. All equipment commonality with the restructured HEAO was a design goal.

B. Electrical Power Requirements

Electrical power requirements for the LST Phase A update are primarily the same as those identified in the Phase A report. No significant changes were identified. Since the OTA and SI requirements have not been finalized as of this date, the estimates used in the earlier analysis were retained.

Based on Phase A thermal analyses of the SI structure, power assumed to be required for instrument heater during the time when instruments were turned off may not be necessary if graphite-epoxy structure material is utilized. Hence, some reduction in power requirements could probably be allowed. However, there is an increasing desire for image tube elements to operate at cooler temperatures, and the entire instrument selection will likely change in Phase B, all of which will alter the power requirements. It was therefore decided to leave the SI requirements as stated in the Phase A report.

Minor power changes were defined in the EPS, ACS, and RCS subsystem; but their magnitude was not sufficient to change the total power requirements. The new power requirements are listed in Table IX-1.

The EPS design capability remains the same with an orbital average power of 1500 W end of mission capability. The EPS is rated for a peak power of 3100 W to limit the discharge rate to half the rating of the combined batteries. The average LST requirement is 1249 W with a peak of 1886 W, giving a design margin of 251 W and 1214 W, respectively. Further discussion and analysis of the loads can be found in Volume V of the Phase A report.

TABLE IX-1. SUBSYSTEM POWER REQUIREMENTS

	Average Watts	Peak Watts
Attitude control system	156	205
Communication and DM	88	127
Power system	30	135
OTA	501	819
SI	469	469
RCS	5	131
Subsystem power requirements	1249	1886
Design margin	251	1214
EPS power rating (out)	1500	3100

C. System Description

The LST electrical system is primarily located in the SSM. The EPS and EDS are functionally interdependent; as a system they are self-sustained and provide the necessary power and distribution services to operate the LST. These subsystems are composed of assemblies that house the devices and components required for generating, storing, and conditioning power and for controlling and protecting the distribution networks.

Automatic controls protect the system against possible failure, particular attention being given to avoiding catastrophic conditions that would impair the ability to recover and maintain the vehicle. System operations are controlled from the ground by way of the command link. Power management can be controlled by way of this link in response to the status and diagnostic data from telemetry.

The configuration of the electrical system was influenced by the configuration and electrical requirements of the LST, the mission duration, reliability requirements, and cost and maintenance considerations. Except for special scientific items, existing technology was used throughout. Space proven assemblies have been used wherever it was determined that such have adequate capacity and electrical characteristics. Such programs as HEAO, OAO, Skylab, ATM and Air Force programs were drawn upon for existing subassemblies for the LST power system.

A simplified block diagram of the SSM electrical system is given in Figure IX-1. The function, description, and rationale for selection of each of the major subassemblies are given in the Phase A report. Only changes from that baseline will be discussed in this update.

A tabulation of the major components comprising the updated electrical power system along with weight and size data is given in Table IX-2. The updated system changes from the reference design are identified below.

1. Rollup solar array with 2-ohm-cm cells.

2. Battery quantity increased by one and changed to HEAO type.

3. Regulator capacity changed to 800 W and quantity decreased to four.

4. Distribution redundancy and cabling reduced.

Discussion and rationale for each of these changes are covered in the following paragraphs.

1. Solar Array. The flexible rollup arrays were a strong contender during the Phase A study. Because of stowage constraints on the alternate Titan launch the reference design was chosen in favor of the rollup type. The removal of the alternate launch vehicle capability resulted in the choice of a different array during this update.

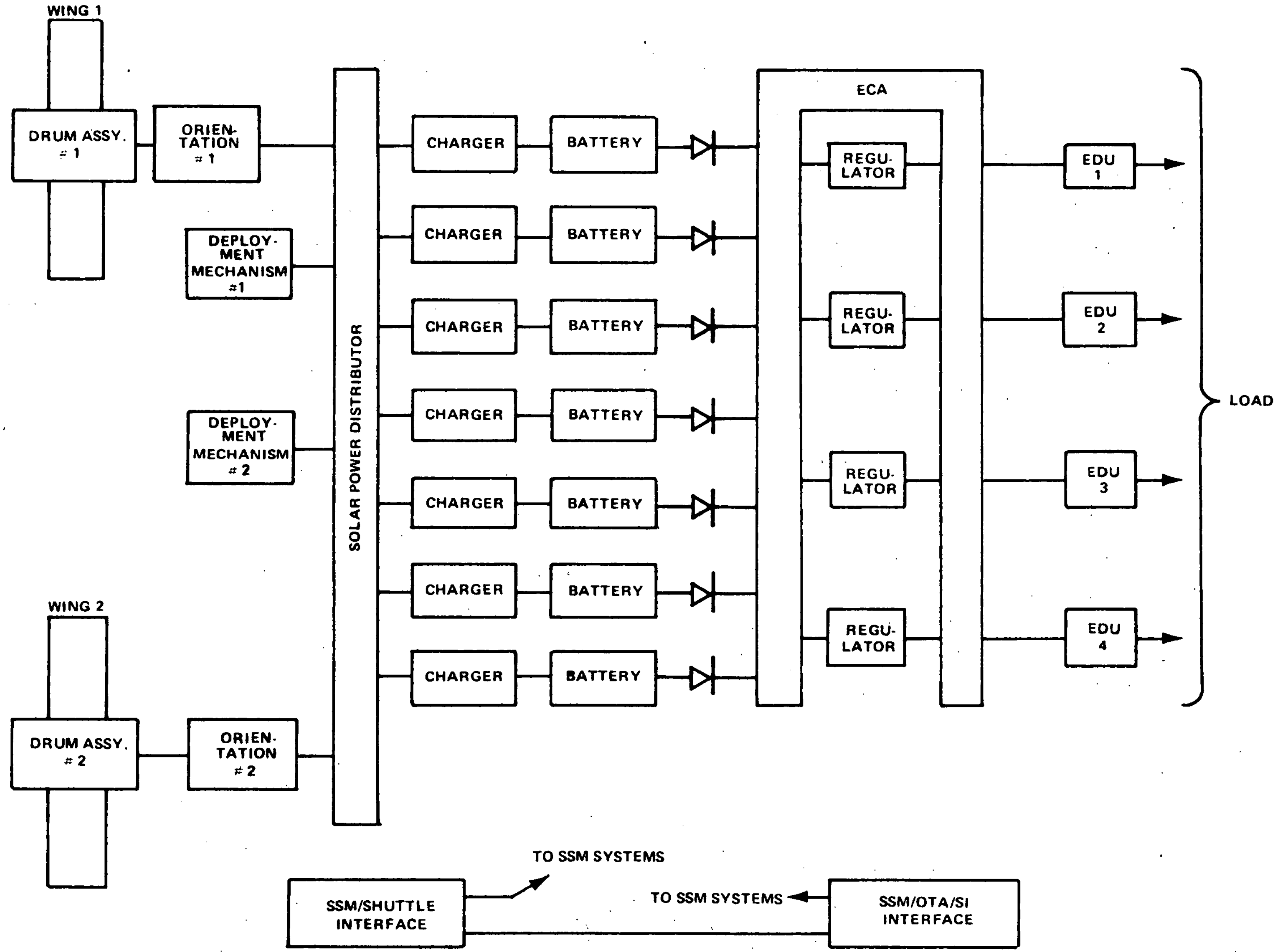


Figure IX-1. LST electrical system block diagram.

TABLE IX-2. LST PHASE A ELECTRICAL SYSTEM EQUIPMENT LIST

Components	Quantity	Size		Mass			
				Unit		Total	
		mm	(in.)	kg	(lb)	kg	(lb)
Batteries	7	419 x 508 x 179	(16 1/2 x 20 x 7)	33.6	(74.0)	235.0	(518.0)
Chargers	7	305 x 203 x 153	(12 x 8 x 6)	6.8	(15.0)	47.6	(105.0)
Regulators	4	305 x 203 x 127	(12 x 8 x 5)	4.5	(10.0)	18.1	(40.0)
Solar power distr.	1	356 x 203 x 178	(14 x 8 x 7)	9.1	(20.0)	9.1	(20.0)
Electrical distr. unit	4	152 x 102 x 76	(6 x 4 x 3)	1.1	(2.5)	4.5	(10.0)
Solar array							
Panels	4	167 x 5587	(66 x 220)	12.0	(26.5)	48.1	(106.0)
Storage drums	2			14.8	(32.7)	29.7	(65.4)
Orientation mech. and electronics	2			33.6	(74.0)	67.2	(148.0)
Array cushion and drive	2			1.5	(3.2)	2.9	(6.4)
Electronic control assembly	1	356 x 203 x 203	(14 x 8 x 8)	7.3	(16.0)	7.3	(16.0)
Cabling and connectors						90.7	(200.0)
		Subtotal				560.0	(1235.0)
Caution and warning cable	1					3.6	(8.0)
Cable and test accessories						15.9	(35.0)
Lighting						9.1	(20.0)
		Subtotal				28.6	(63.0)
		Total				588.6	(1298.0)

A comparison of the reference design with two rollup arrays (Figs. IX-2 and IX-3) is given in Table IX-3. The rollup arrays offer lower weight, more compact stowage, better adaptation to on-orbit maintenance, and most important to this maintenance concept, ease of retraction. Of the two rollup concepts considered the FRUSA, or one of its type, was selected for the update configuration. Although the complexity of the FRUSA is greater than that of the alternate, its flight experience, shorter boom length with resultant higher natural frequency, and compatibility with mounting requirements for the antenna of the communications system determined its selection. A more detailed description of both arrays can be found in Volume V of the Phase A report.

Ground return at the end of 2 1/2 years suggests a change of the arrays at this interval rather than a 5-year interval as baselined in the Phase A design effort. This renewal rate decreases the array size, makes use of a higher efficiency cell possible, and improves the confidence level of end-of-life performance. The choice of a 2-ohm-cm solar cell was made for this mission length because of its early life efficiency. As can be seen from Figure IX-4 the crossover point for the 2- and 10-ohm-cm cells is 3 years. Prior to this time the 2-ohm-cm cell had a higher output efficiency and at the 2 1/2-year end of life was 1.4-percent better than the 10-ohm-cm cell. Regardless of array choice, changing of the array each time the LST is returned for maintenance and the use of 2-ohm-cm cells are recommended.

Cell characteristics and array size and mass data are given in Tables IX-4 and IX-5, respectively. Overall effective array area was reduced by 2.23 m^2 as a result of reduced life expectancy and lower degradation losses. Figures IX-5 and IX-6 show the selected array in the deployed and stowed positions, respectively.

Cost of rollup type arrays averages 15 to 20 percent higher than the conventional type of the reference design. The cost difference is expected to be lowered by the time of the production phase of LST as a result of wider use of the rollup configuration. Based on a 2 1/2-year replacement rate, the use of 2-ohm-cm cells on either array type and the FRUSA initial cost being 15 percent greater than the conventional array, overall array program cost for the 15-year SSM life would be approximately 30 percent less with the FRUSA. This apparent saving is the result of only the solar cell blanket of the rollup array being replaced each refurbishment period. The blanket cost is roughly 45 percent of the total array cost. The drum and orientation mechanism would not need replacing; however, two complete arrays are recommended to facilitate fast turn around, the spare drum being available for array replacement during each mission. The cost of the spare drum is included in the previous cost statement.

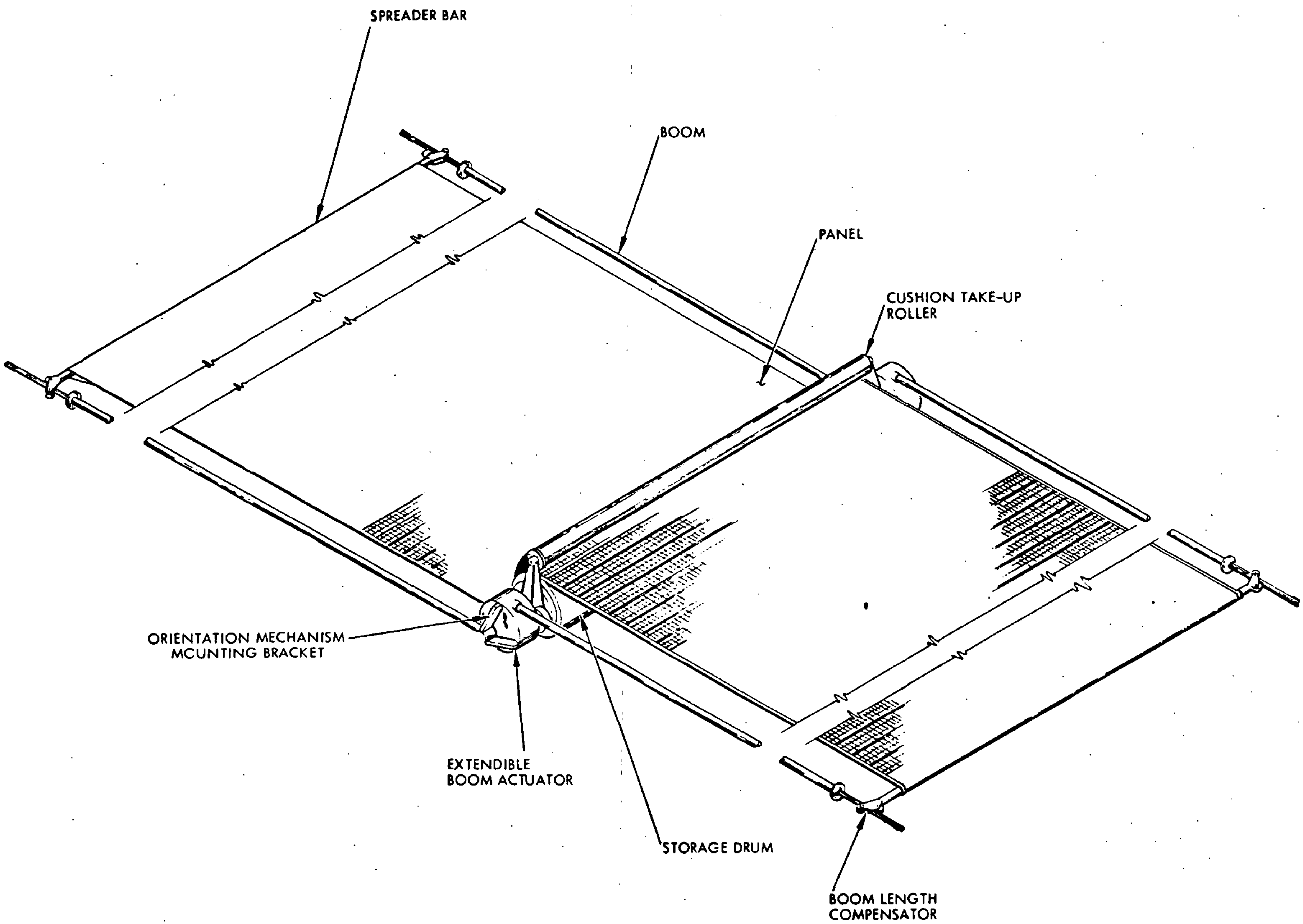


Figure IX-2. Flexible roll-up solar array, alternate no. 1 (FRUSA)

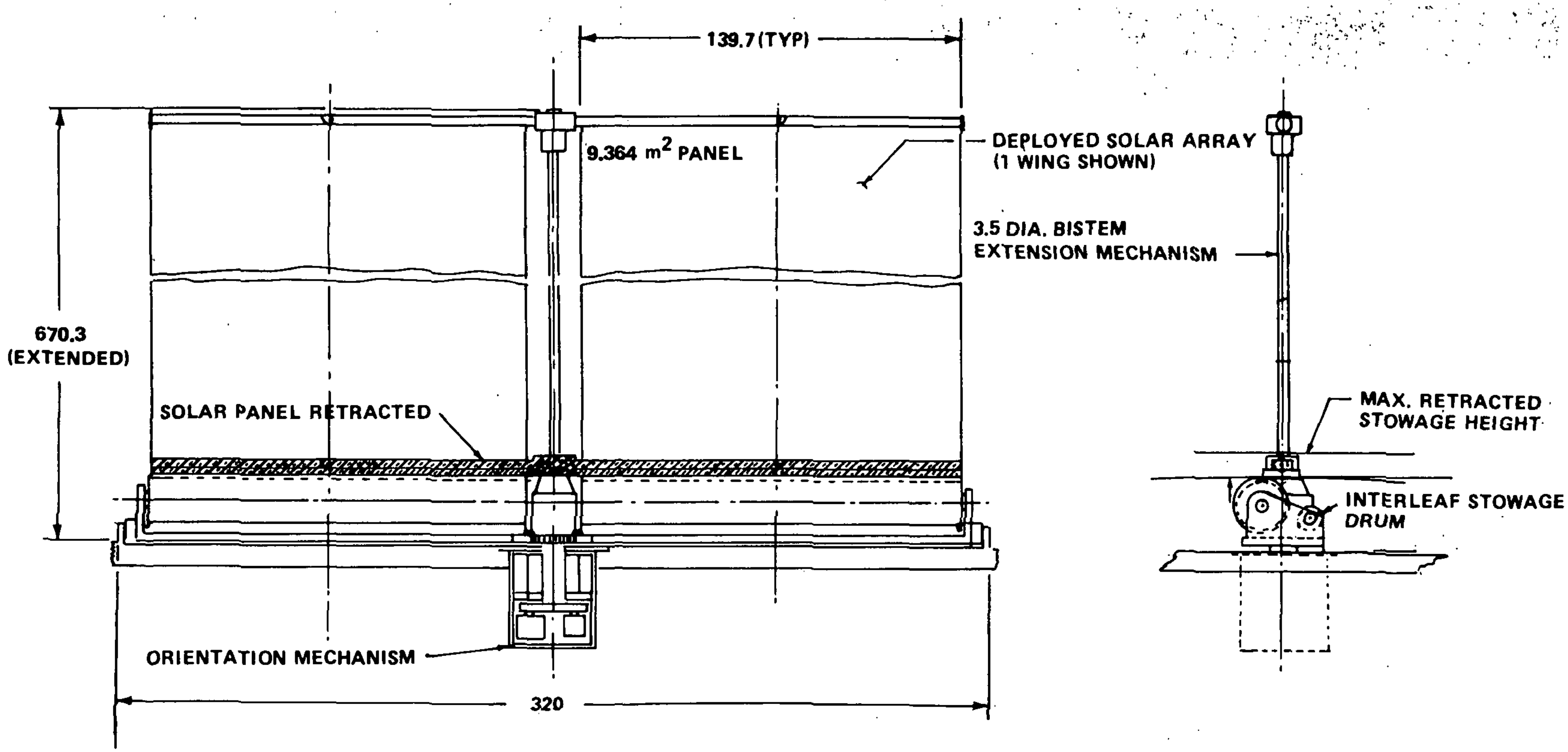


Figure IX-3. Flexible roll-up solar array alternate no. 2 (FRUSA)

TABLE IX-3. ARRAY SELECTION

Type	Advantages	Disadvantages
Reference design concept (rigid foldout)	Proven technology Proven fab. methods Presently lowest cost High natural freq. (0.6 - 1.0 Hz)	High mass Constrained to wrap around stowage Complex retraction Difficult to maintain in space
Flexible roll-up solar array (FRUSA)[a]	Low mass Compact stowage Flight tested Easier retraction Better adapted to in-space maintenance Offers antenna mounting for communication system Shortest boom length	Most complex design Low natural freq. (0.4 - 0.1 Hz) Must be retracted for docking Presently 15/20 percent higher cost
Alternate roll-up array	Lowest mass Less complex than FRUSA	New design Difficult to mount communication antenna on array Longer boom length Lowest natural freq.

a. Selected array for Phase A update.

2. Energy Storage Subsystem. The energy storage subsystem must store energy in the sunlight periods of the orbit and furnish the LST electrical load requirements during the occultation periods. Solar array power must be supplemented occasionally by the energy storage system during off-sun orientation. The system also provides means for conditioning and controlling energy storage.

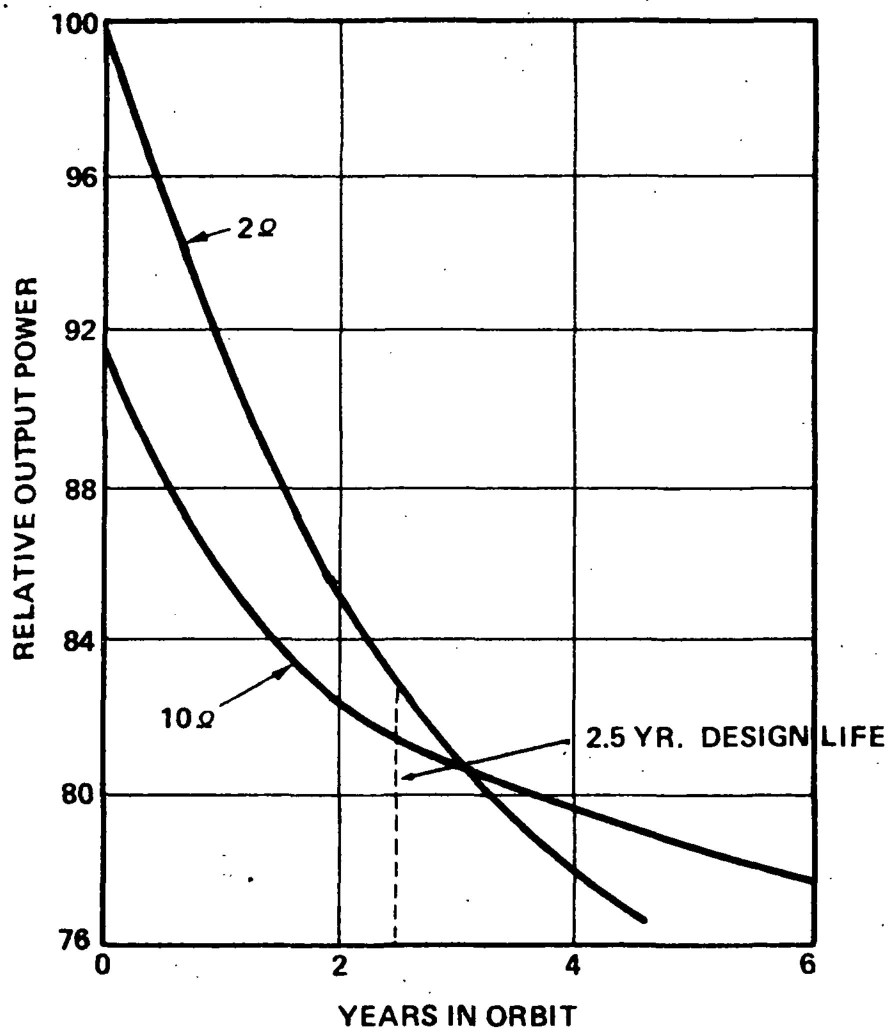


Figure IX-4. Solar array degradation.

TABLE IX-4. SOLAR CELL CHARACTERISTICS

Type	Silicon N/P
Size	2 x 4 x 0.031 cm
Base resistivity	2-ohm-cm
Contacts	Ag-Ti
Effective area, min	7.7 cm^2
Open circuit voltage	587 mV
Short circuit current	272 mA
Max power voltage	474 mV
Max power current	245 mA
Efficiency, nominal	11.5 percent

TABLE IX-5. SOLAR ARRAY CHARACTERISTICS

Stowed dimensions		
Height	698.5 mm	(27.5 in.)
Length	2951.5 mm	(116.2 in.)
Width	297.2 mm	(11.7 in.)
Deployed parameters		
Panel size	1676 x 5587 mm	(5 1/2 x 18 1/3 ft)
Total array area	36.79 mm^2	(396 sq ft^2)
Total array and orientation mass	147.9 kg	(326 lb)

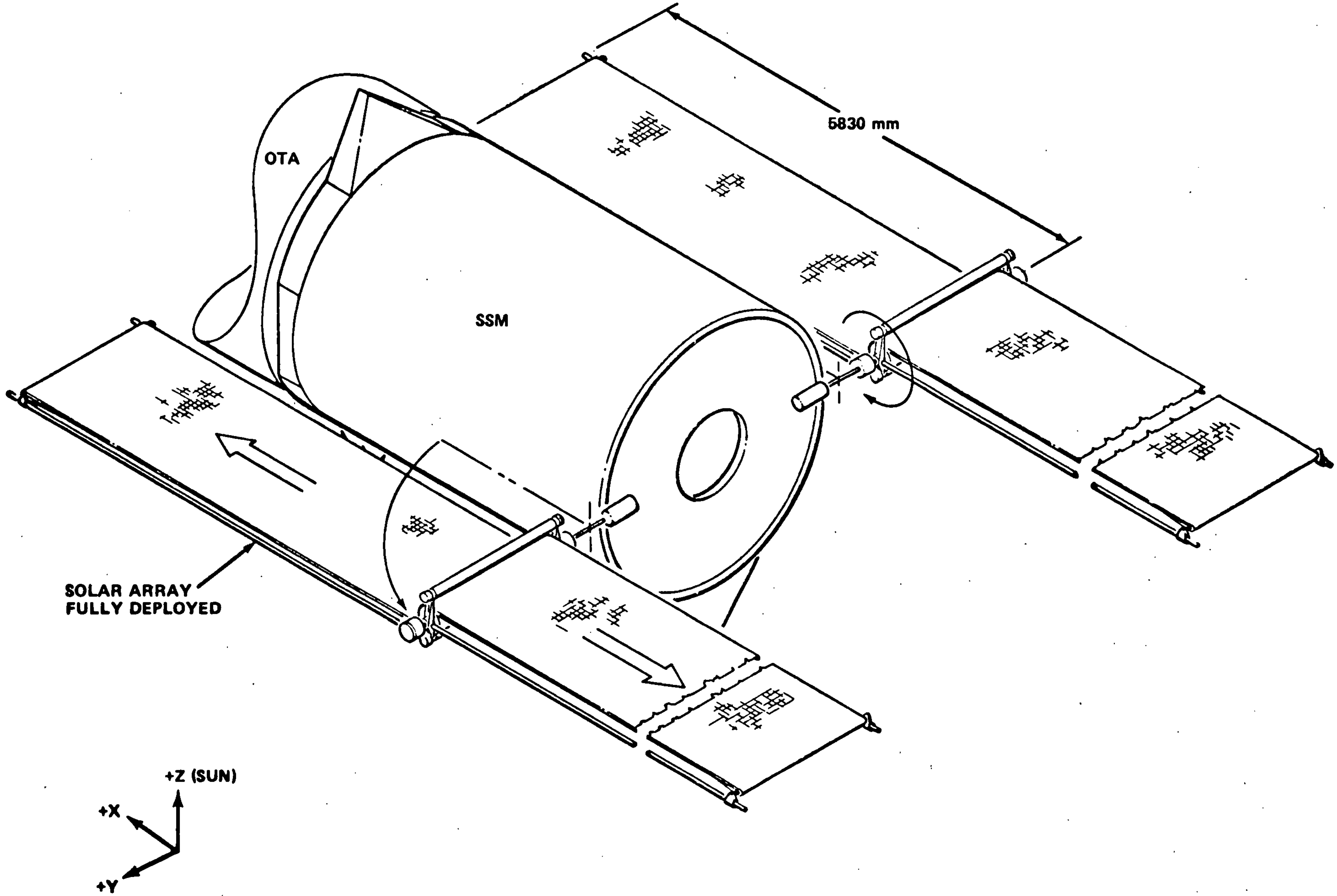


Figure IX-5. Flexible roll-up solar array (deployed)

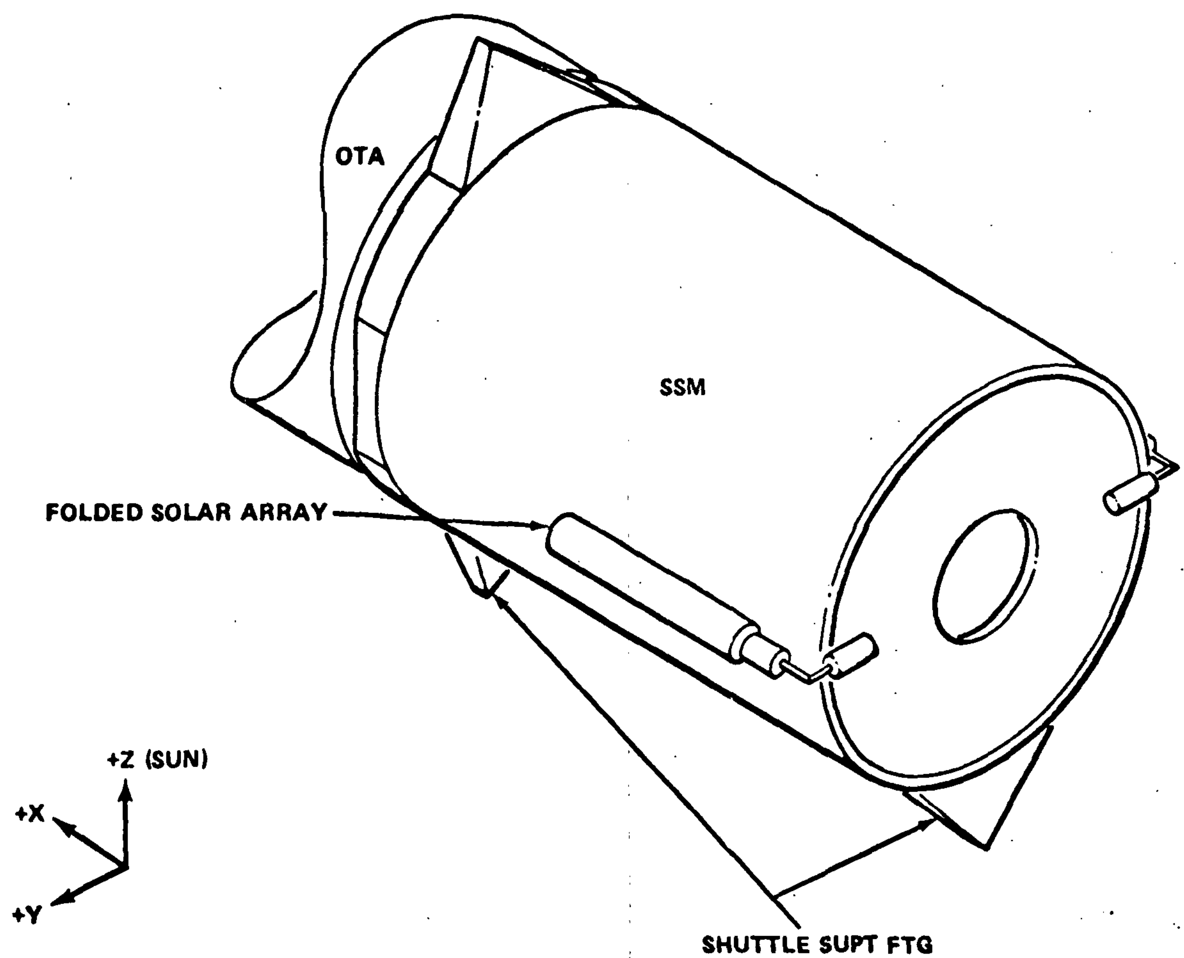


Figure IX-6. Flexible roll-up solar array (stowed)

Loads, efficiency, duration of occultation, temperature, and life are significant drivers in the selecting and sizing of the energy storage subsystem. Based on rated power of 1500 W for a maximum occultation of 0.592 hours, an energy capacity of 888 Wh is required. The subsystem efficiency of 86.5 percent requires the batteries to deliver 1027 Wh of energy each occultation period. Battery life is dependent upon temperature and depth of discharge (DOD). Figure IX-7 shows the DOD that can be used to predict cycle life, with confidence, at the temperature predicted for the LST. The 2 1/2-year life required before ground maintenance limits the DOD to 16 percent. To accommodate the 1027 Wh required at a 16-percent depth of discharge seven batteries will be required. The energy storage subsystem is sized for a 2 1/2-year life with no on-orbit maintenance.

To establish commonality with other programs and reduce cost, a Mariner battery of the type planned for the Restructured HEAO was selected. Individual battery mass increases almost 50 percent and volume increases by 1.76×10^4 cm^3 per battery. Existing design and commonality with other programs should lower cost from the reference design. Characteristics of the battery are tabulated in Table IX-6.

It is assumed that battery replacement would be included in any on-orbit maintenance operation. Based on a maintenance visit within 1 1/2 years of launch, the DOD could be increased to 28.5 percent. The result would be a four-battery system with a total mass reduction of 121.2 kg from the seven-battery configuration.

3. Regulators. The Phase A design utilized six regulators. The system capability required the capacity of fewer than three of these units. The remainder was required as part of the redundancy scheme. By increasing the capacity of each regulator to 800 W, the same philosophy was maintained with only two units required to operate the system. Some reliability was lost going to a two-out-of-four from a three-out-of-six configuration. The reduction in quantity of units and system complexity was considered great enough to offset the small loss in reliability.

4. Distribution. During this study an effort was made to reduce cost and complexity without substantial loss of realibility. Several configurations of the EPS and EDS subsystems with various redundancy combinations were considered. Realibility numbers were generated for each approach and the system shown in Figure IX-1 was selected. Comparison of Figure IX-1 with the block diagram of the Phase A design will reveal a reduction in redundancy, primarily in the distribution area. Since these items have the lowest failure

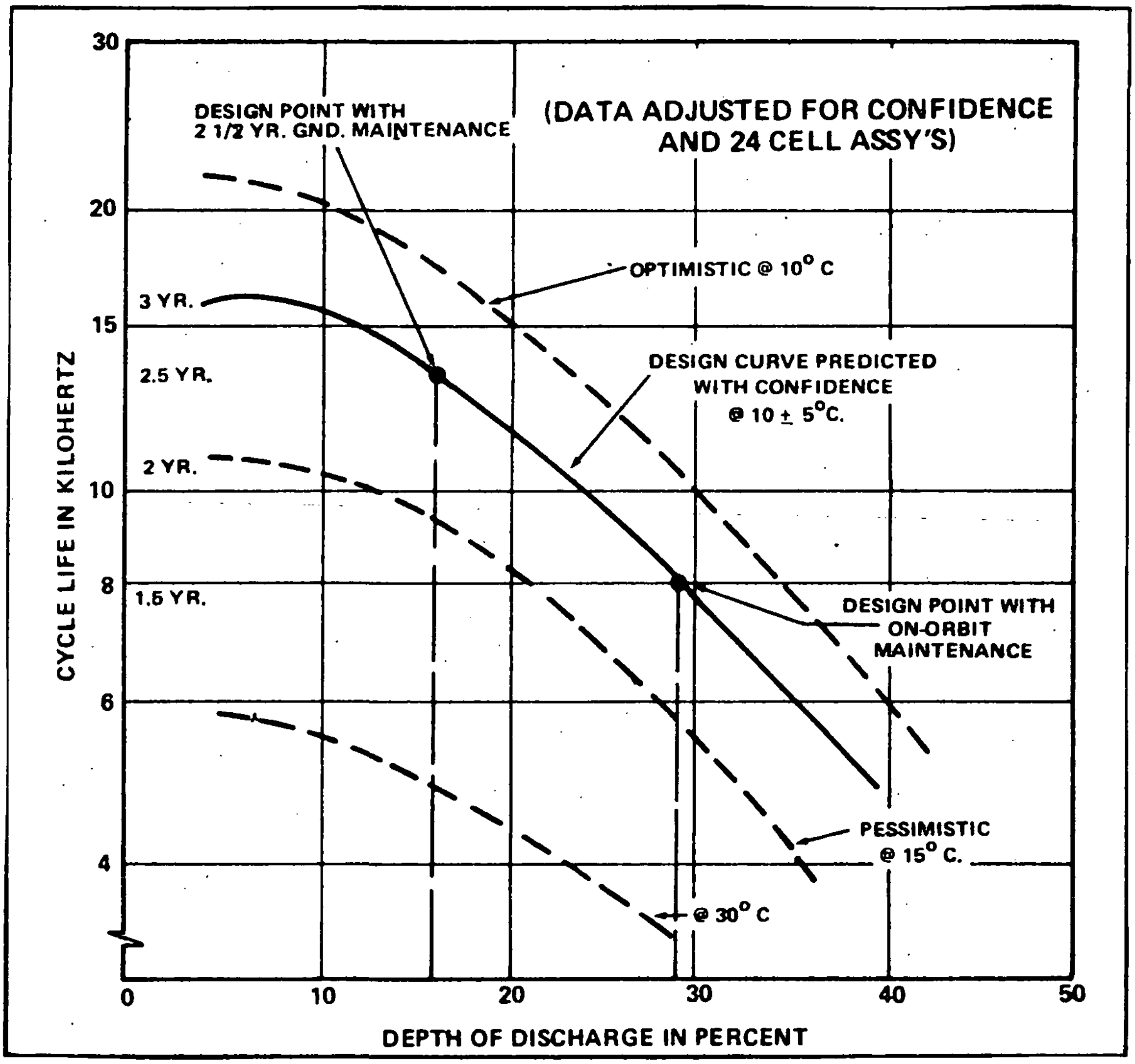


Figure IX-7. Battery life predictions.

TABLE IX-6. ENERGY STORAGE SUBSYSTEM CHARACTERISTICS AND RATINGS

Characteristics	Ratings
<u>Batteries</u>	
Number of assemblies	7
Type	NiCd
Cell rating	30 Ah
Temperature	10° ± 5° C
Average DOD	16 percent
<u>Charger Assemblies</u>	
Number of assemblies	7
Type	Stepdown converter
Max recharger rating	4700 W
Max charge voltage	36 V
<u>Total Subsystem</u>	
Energy capacity	6300 Wh
Average power	3100 W
Max current	210 A
Output voltage	24 to 31 Vdc
Input voltage	38 to 88 Vdc
Efficiency	73.5 percent
Life	2.5 yr
Mass	282.7 kg (623 lb)
Volume	3.29 x 10^5 cm^3 (11.6 ft^3)

rate of this system, they affect system realiability the least. Also these items have built-in redundancy and even though a simpler flow appears to exist, dual paths are present from source to load in all cases.

The solar power distribution has been repackaged in this configuration. It maintains all of the capability of the two SPDs of the reference design. The quantity of assemblies was reduced in an effort to reduce complexity and overall system cost. The remaining distribution units remain unchanged. Their dual circuit capability retains the single-point failure protection. Thus, the total number of units can be reduced while retaining system redundancy without sacrificing a great loss in reliability.

Cable quantity and mass are reduced in two areas (1) distribution unit reduction and (2) deletion of inactive equipment required for on-orbit maintenance. Cable mass was reduced to 90.7 kg and inactive equipment reduced by approximately 22.7 kg. Total system mass was decreased by 97.6 kg.

D. Off-Sun Pointing Capability

The guidelines and requirements for the Phase A update call for an off-sun pointing capability of a maximum of three consecutive orbits not to occur more frequently than once per day. The EPS capability during off-sun pointing for polarimetry experiments is covered in depth in the EPS section of Volume V of the LST Phase A report.

The capability in the updated design has increased, as shown in Figure IX-8. This increase comes as a result of increased battery capacity and as shown can accommodate three orbits at any off-sun angle if the roll is limited to ±45 deg. To accommodate off-sun pointing a DOD of 50 percent is permitted. Frequent occurrences at this DOD would weaken the batteries and reduce their life expectancy. A 2 1/2-year life expectancy with reasonable confidence could not be obtained if off-sun pointing for polarimetry were required on a daily basis. Off-sun pointing with no roll restrictions would not limit the solar array performance.

Off-sun pointing requirements must be clarified. Polarimetry requirements and frequency of occurrence should be studied during the Phase B study to determine their impact on the EPS.

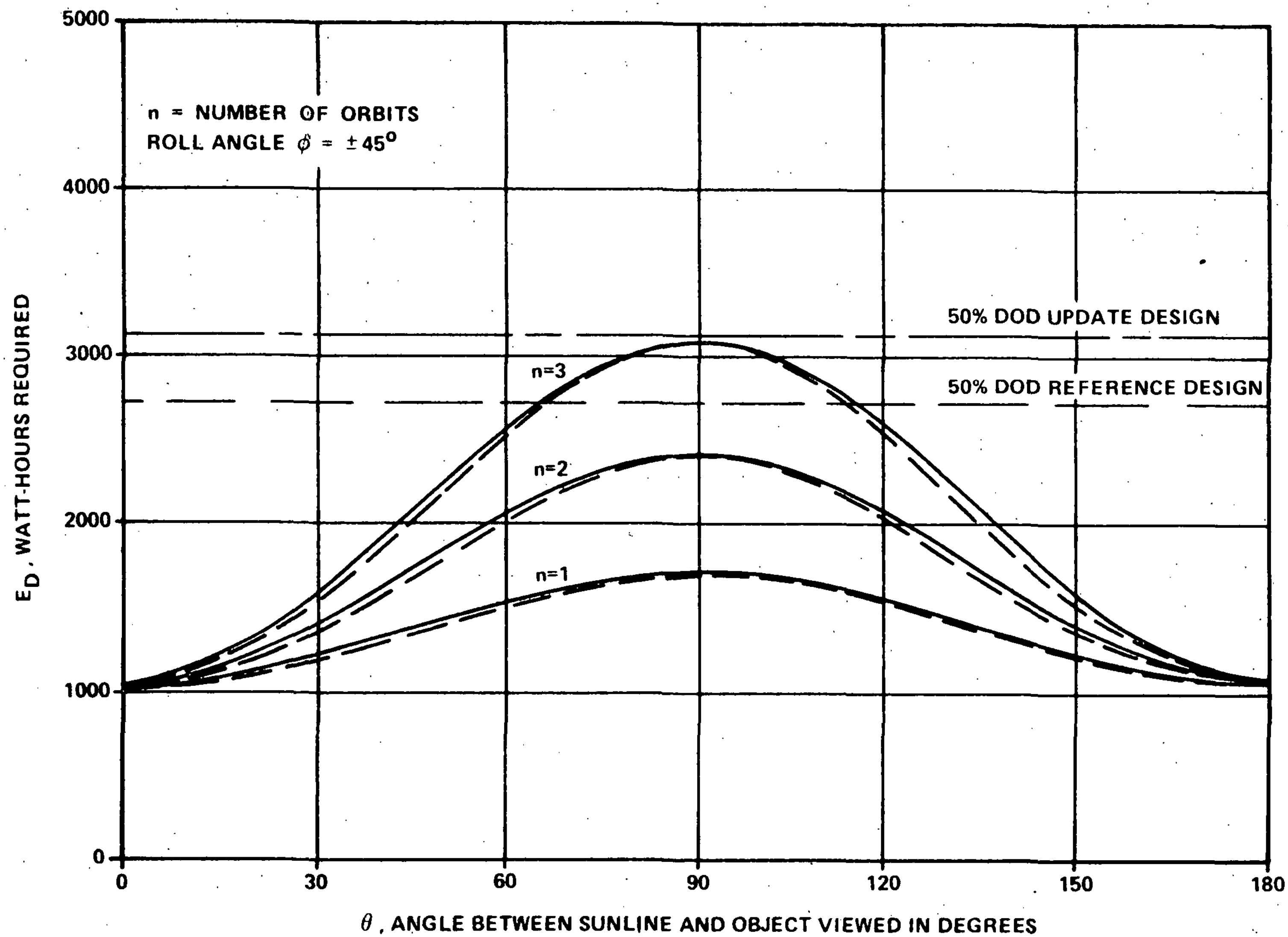


Figure IX-8. Energy required with $\phi = \pm 45$ degrees and end of life solar array.

E. Maintenance Mode Effects on Electrical System and Support Equipment

The major effort in this portion of the Phase A update was directed toward the ground-return maintenance mode. Some investigation was also made into on-orbit maintenance, either EVA or by way of manipulator. The impacts of these concepts on the electrical system are compared in Table IX-7. Note that manipulator maintenance has a greater impact on all areas except ESE. The ground maintenance is less complex than any of the maintenance modes considered. The ESE is reduced by a larger percent than other areas, both in amount required and complexity.

The biggest impact on the design of the electrical system by the maintenance modes is in the area of interfaces and connectors. Table IX-8 identifies some of the problems associated with on-orbit maintenance. In the EVA case it was assumed that all connections would be made by an astronaut, while in the manipulator concept all interfaces were assumed broken by automatic seating connectors operated by the manipulator.

F. Summary and Conclusions

The basic electrical power system is the same as that of the Phase A design. Component selection changes and reduction in redundancy comprise the only changes. The majority of these changes resulted from the effort to reduce cost and promote commonality with other programs.

Rollup array selection was allowed by stowage constraint removal with the deletion of Titan launch requirements. Battery selection was changed for commonality with the restructured HEAO. No significant changes in power requirements were identified in the update effort. Battery quantity, solar cell selection, and array size were changed to be compatible with life requirements dictated by the initial refurbishment interval. The remaining changes were recommended to reduce overall system complexity and cost.

Maintenance mode (that is, EVA, ground maintenance, and so forth) was not a driver on the selection of electrical system components. Interface complexity, cable and connector weights and volume, and electrical support equipment are influenced by maintenance concept. However, with the exception of connector problems with a manipulator, these items do not appear to be a driver on maintenance mode selection.

Selective reduction in redundancy appears feasible without substantial loss in reliability. A corresponding decrease in complexity and cost should accompany the redundancy reduction.

TABLE IX-7. EFFECT OF MAINTENANCE MODE CONCEPTS ON LST ELECTRICAL SYSTEM AND SUPPORT EQUIPMENT

Program Factor/Equipment	Relative Ratings		
	Phase A Design Concept	Manipulator Maintenance Concept	Ground-Return Concept
Design and development			
Solar array area	1.00	1.00	1.00
Design complexity	1.00	1.03	0.97
Weight	1.00	1.03	0.97
EPS equipment weight	1.00	1.06	0.99
Volume	1.00	1.15	0.97
Design complexity	1.00	1.10	0.98
Distribution and interfaces wt.	1.00	1.20	0.93
Design complexity	1.00	1.12	0.94
Volume	1.00	1.15	0.95
Testing Required			
Develop/accept/DVT	1.00	1.12	0.92
Elec GSE/facilities complexity	1.00	0.93	0.87
Manufacturing (elec)	1.00	1.04	0.96
Flight ESE			
Quantity	1.00	0.92	0.85
Complexity	1.00	1.05	0.80

NOTE: Complexity numbers — 1 is used as a reference; percentage > 1 represents magnitude of complexity greater than reference.

TABLE IX-8. MAINTENANCE MODE IMPACT ON INTERFACES

Modular Concept With EVA	Manipulator	Ground Maintenance
High confidence in zero g connector	No existing design adequate	Existing design adequate for all interfaces
Require pigtail from tray to connector	Technology adequate, but would require an R&D effort (probably not extensive)	Least amount of connectors
Increased weight and number of interfaces	RF connectors biggest problem	Less cable and connector weight
Other connectors not adaptive to astronaut use	Highest weight	
More complex	Most complex	Least complex

CHAPTER X. COMMUNICATIONS AND DATA HANDLING

A. Guidelines and Constraints

The basic requirements for the Communications and Data Handling (C&DH) System are identical to those defined in the LST Phase A Report. During the Phase A update the C&DH System was examined to determine impacts of ground-return maintenance, manipulator maintenance, extravehicular activity (EVA) maintenance, 2 1/2-year lifetime prior to first ground return, and cost considerations of reduction in redundancy. This section presents discussions of the impacts resulting from these new concepts. In addition, trade studies that consider the impacts of rollup solar arrays upon the antenna subsystem and examine revised antenna switching techniques are presented.

B. Impacts of New Guidelines

This section presents discussions of the impacts upon the C&DH System resulting from the new guidelines mentioned above.

1. Maintenance Impacts. The revised maintenance mode has no effect on the functional operation of the C&DH System.

2. Redundancy/Cost Considerations. In an effort to reduce cost, it has been proposed that a reduction in subsystem redundancy be considered for all LST Systems. A reduction in redundancy would also reduce spacecraft system complexity, and in some cases power. However, redundancy should not be removed in areas that are critical to vehicle communications or recovery. Considering the vital role that the C&DH System plays in the commanding of the vehicle and the acquisition of subsystem status information, it is recommended that the C&DH System retain full redundancy. The reliability of the C&DH System is critical to the LST mission.

C. System Trade Studies

1. Impacts of Rollup Solar Arrays upon the Antenna Subsystem. The removal of the alternate Titan vehicle for LST launch resulted in the choice of flexible rollup solar arrays as the reference design power source. Since the Phase A reference design has the antennas of the C&DH System located on the ends of the solar arrays, the two candidate rollup arrays were examined to determine if the antennas could be similarily located on these arrays.

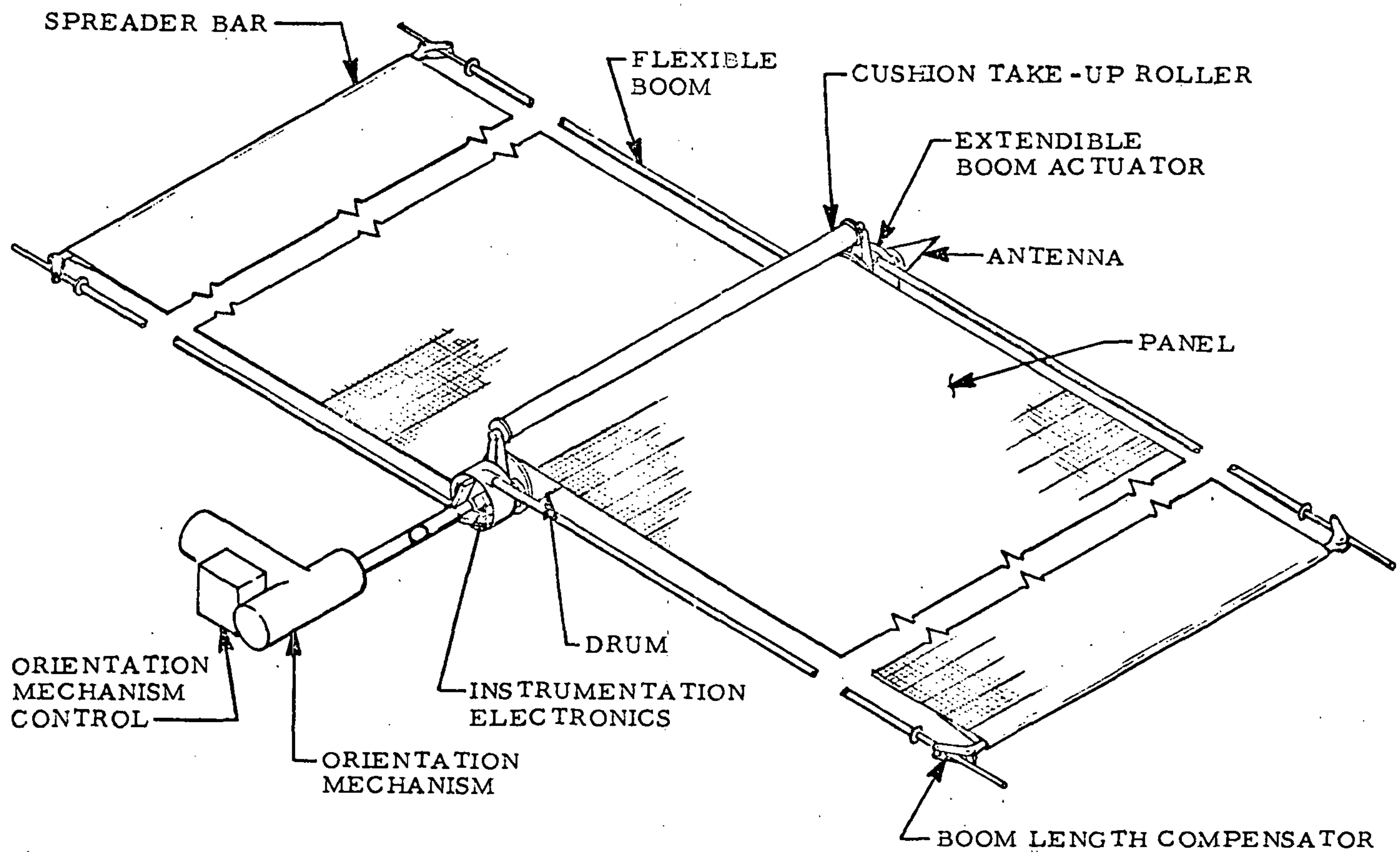
Analyses of the two arrays, Figures X-1 and X-2, show that either system will allow antennas to be mounted as specified in the Phase A LST report. It was also determined that both array systems retract and are stored in a manner that will allow limited communications when the arrays are in a stored mode. Further analysis in this area should be performed during Phase B.

The critical factor of the two rollup array systems from the standpoint of C&DH is the method of array rollup and deployment. Alternative 1 utilizes a rigid cylindrical boom holding a drum that houses the rolled up arrays. The rigid boom is swung from its stored position alongside the LST body to a position perpendicular to the LST body. At this time the arrays are deployed from the drum by flexible rods which are extended by the extendable rod actuator. By utilizing the rigid cylindrical boom, a rigid or semirigid coaxial cable and connection can be maintained between the antenna and the transponder located in the body of the LST.

Solar array Alternative 2 utilizes a single flexible boom (bistem extension mechanism) to deploy the stored arrays from drums located alongside the LST. To maintain connection between the antenna and transponder in this situation, a flexible coaxial cable would have to be employed along with a cable rollup mechanism containing an RF rotary joint. This combination of RF rotary joint and the required flexible coaxial cable would mean an approximate 6-dB penalty in signal-to-noise ratio over the configuration of Alternative 1. Thus to avoid this 6-dB penalty and complicated cable rollup mechanism, Alternative 1, also known as the FRUSA (flexible, rollup solar array), was selected as the preferred solar array from the standpoint of C&DH.

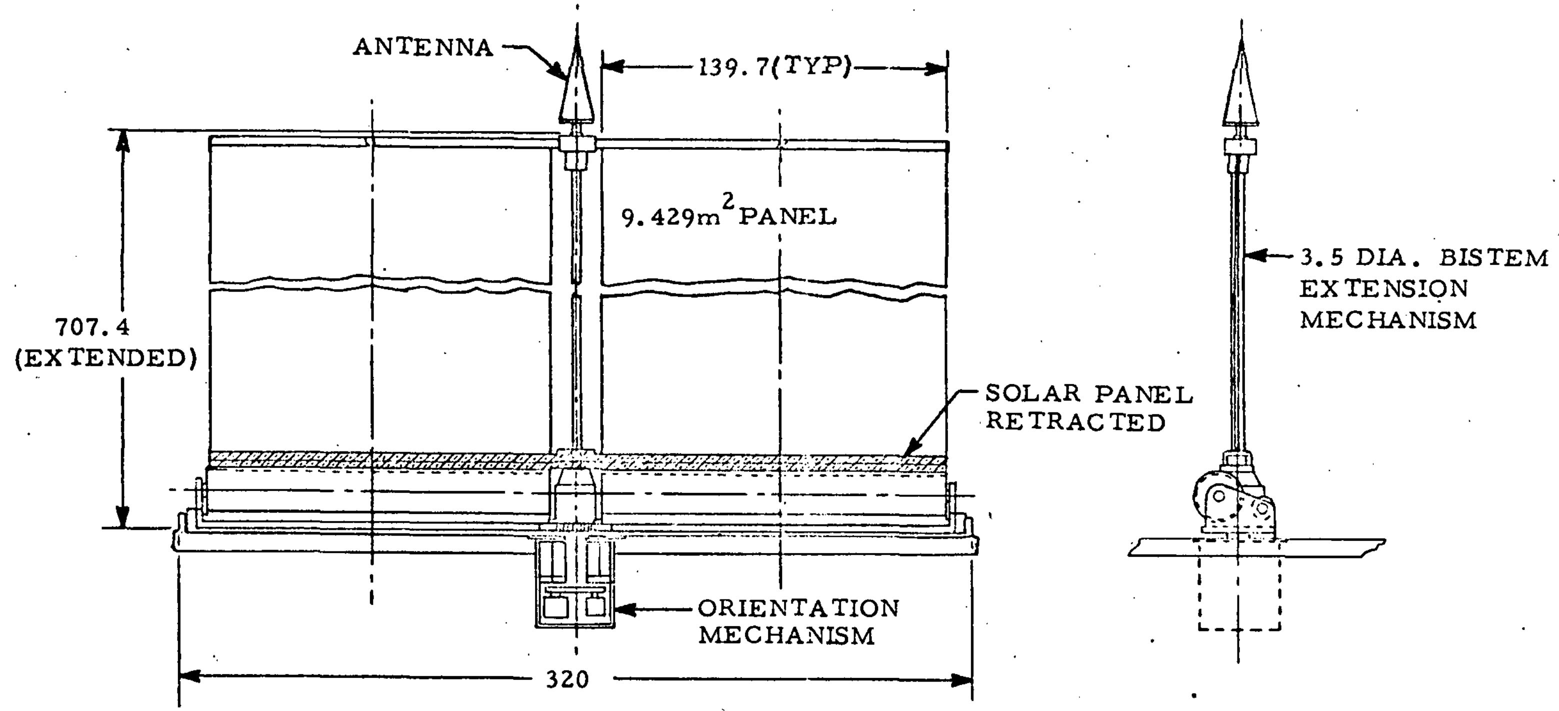
2. Antenna Switching Techniques. The LST Phase A Communications System has been updated to provide increased reliability and optimum system performance. Four candidate systems are presented that will allow either of the circularly polarized, solar-array-mounted antennas to be utilized for the transmission of both scientific data and engineering data. Similarly, either antenna can be used for the reception of ground commands. Each of the candidate systems contains the ERTS unified S-band transponder either with or without modification, depending upon the particular configuration. All configurations contain both a PM and an FM capability.

Figure X-3 is a functional block diagram of communications system Alternative 1. Antenna switching is utilized with six coaxial RF switches (four external to the transponder and two internal) and two hybrid rings. Two different S-band transponders are needed here. One is a modified ERTS with FM capability for the scientific data and the other is an ERTS transponder with PM capability for the engineering data. The particular type of data handled by each transponder is shown in Figure X-3.



- SEMI-RIGID COAXIAL CABLE MAY BE EMPLOYED.
- CABLE ROLL MECHANISM IS NOT REQUIRED.
- AN RF ROTARY JOINT IS NOT REQUIRED.
- COMMUNICATION IS POSSIBLE IN THE STORED MODE, HOWEVER IT WILL BE ORIENTATION DEPENDENT.

Figure X-1. Flexible roll-up solar array, alternate no. 1 (FRUSA)



- FLEXIBLE COAXIAL CABLE IS REQUIRED SINCE THE CABLE MUST BE REELED IN AS THE SOLAR PANEL ROLLS UP.
- A MECHANISM TO ROLL UP THE CABLE MUST BE INCLUDED. THE MECHANISM MUST INCLUDE AN RF ROTARY JOINT.
- COMMUNICATION IS POSSIBLE IN THE STORED MODE, HOWEVER IT WILL BE ORIENTATION DEPENDENT.

Figure X-2. Roll-up solar array alternate no. 2.

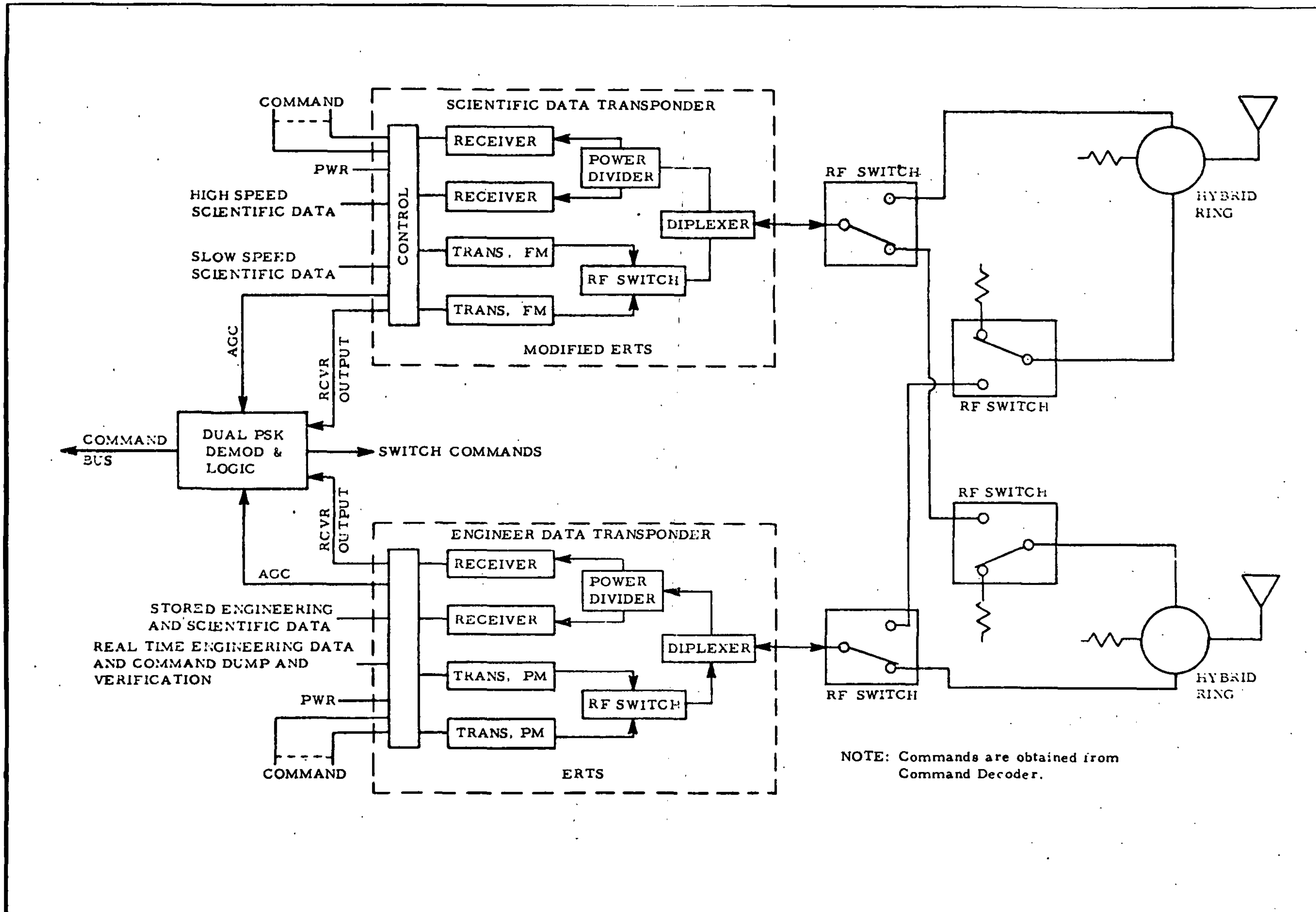


Figure X-3. Communications subsystem alternate no. 1

The following points are noted about Alternative 1:

- Full capability is possible with the failure of a transmitter and/or a receiver.

- Four coaxial RF switches are required external to the transponder. Failure of one of these switches will result in single antenna downlink capability for either the scientific or engineering transponder, depending upon the switch that fails.

- The use of hybrid rings results in 3-dB transmitter loss.

- Full uplink capability is retained with the failure of a single switch and/or a single receiver.

Figure X-4 is a functional block diagram of communications system Alternative 2. This scheme utilizes multicouplers and switching techniques within a downlink data distribution to achieve the required system performance as mentioned earlier. Both transponders are modified ERTS, each having both FM and PM downlink capability.

The transmitters are connected to a single antenna, and data are switched to the transmitter connected to the desired antenna. This technique eliminates the coaxial RF switches that have caused problems in recent NASA flights.

Items to be noted about this alternative are as follows:

- A 20- to 30-MHz frequency separation must be maintained between transmitters because of the use of the multicouplers.

- No coaxial RF switches are required.

- Loss of one transmitter will result in single antenna capability for the kind of information handled by the faulted transmitter.

- Full uplink capability can be maintained with the loss of one receiver.

- Data switching is performed before modulation through the downlink data distributor.

- Both transponders are identical.

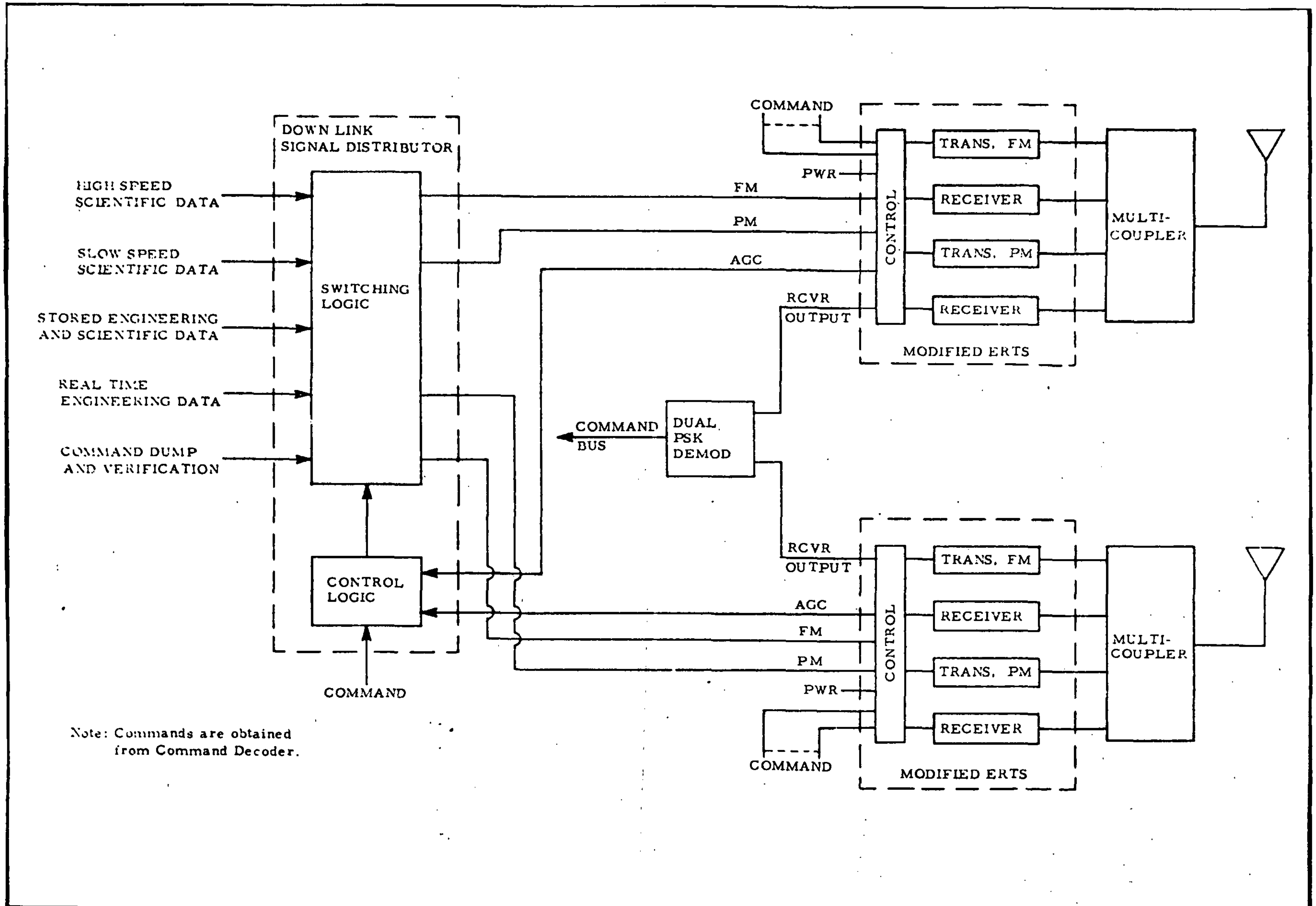


Figure X-4. Communication subsystem, alternate no. 2

Alternative 3 is presented in Figure X-5. Here reversible circulator switches are used external to the transponders in conjunction with multicouplers to attain full system capabilities. The circulator switch was included as an option to the usual RF switch since it requires no mechanical motion during its switching procedure and thus is expected to be more reliable. Figure X-5 shows the circulator switches external to the transponders and another type RF switch inside the transponder. If the circulator switch is chosen for LST use, it will be desirable for all RF switches, both external and internal to the transponder to be circulator switches.

The ERTS transponders used in this alternative are functionally the same as those in Alternative 1, one with FM capability and one with PM capability. As in the previous alternatives, FM is used for scientific data and PM for engineering data. Keypoints about this concept are listed below:

- A 20- to 30-MHz frequency separation must be maintained between transmitters because of the use of the triplexers.

- Full capability is possible with the failure of a transmitter and/or a receiver.

- Two circulator switches are utilized external to the transponder. Failure of one of these switches will result in single antenna downlink capability for either the scientific or engineering transponders, depending upon the switch that fails.

Alternative 4 is presented in Figure X-6. This alternative is a combination of Alternatives 2 and 3. This design has complete flexibility of switching scientific and engineering data to either transponder. It also allows either transponder to use either antenna.

- Full capability is possible with the failure of a transmitter and/or receiver.

- Selective scientific or engineering transmission is possible with the failure of a complete transponder.

- Data switching is performing before modulation through the downlink data distributor.

- A 20- to 30-MHz frequency separation must be maintained between transmitters because of the use of the multicouplers.

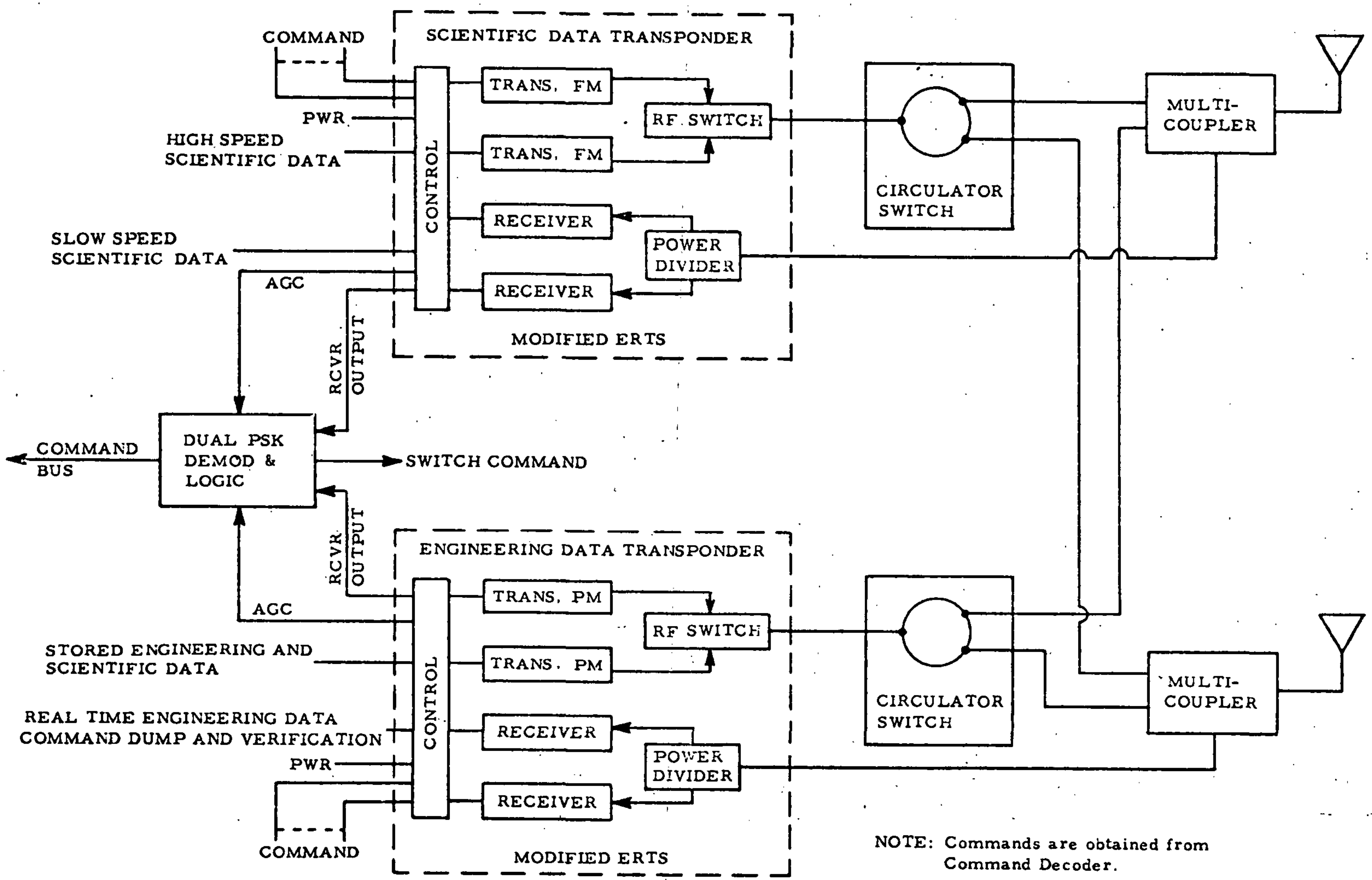


Figure X-5. Communications subsystem alternate no. 3.

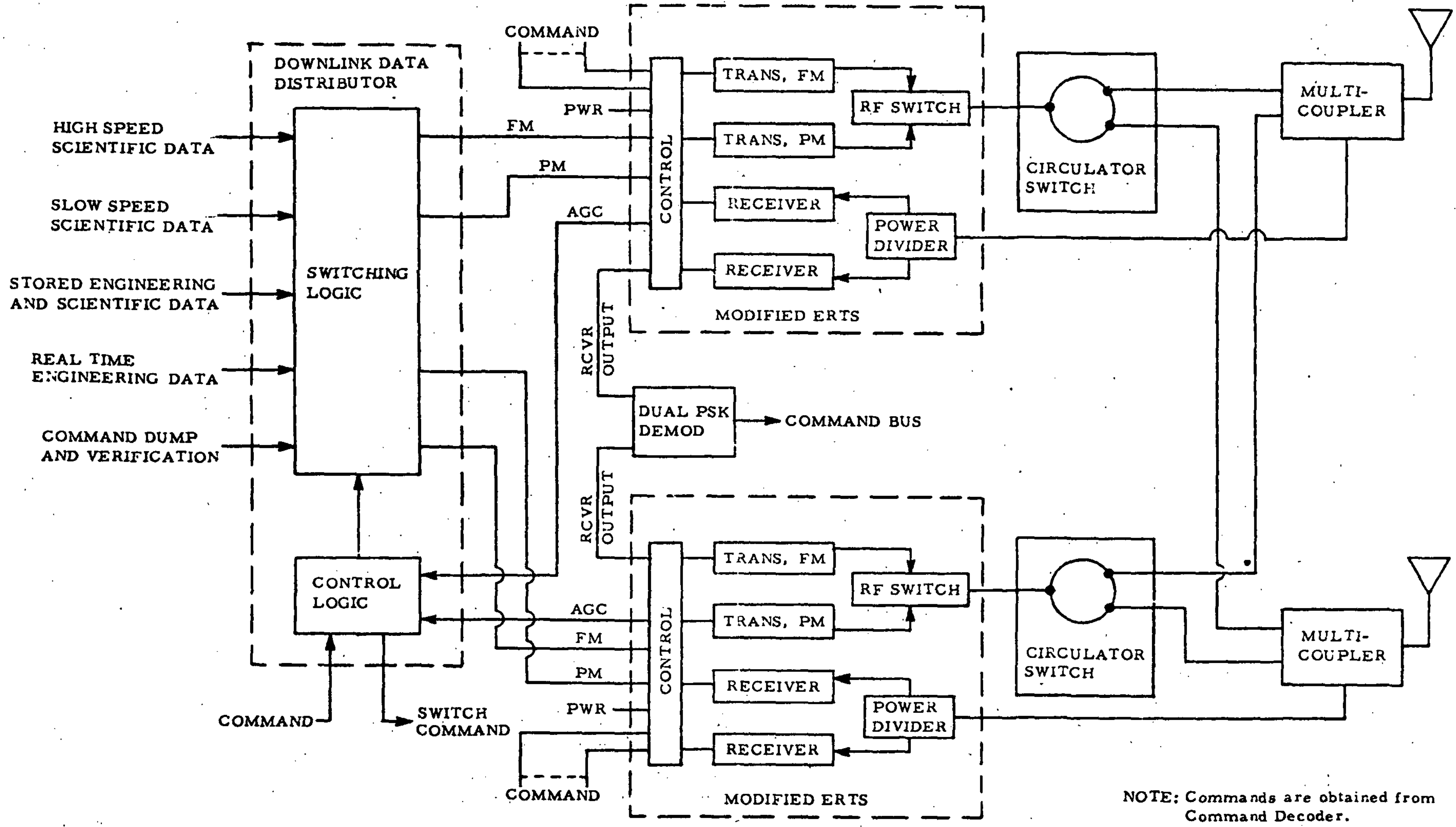


Figure X-6. Communications subsystem alternate no. 4.

- No coaxial RF switches are required.

- Both transponders are identical.

- Two circulator switches are included. Failure of one of these switches will result in reduced downlink capability over one antenna. Full downlink capability is retained over the second antenna.

All communication subsystems alternatives are feasible and practical. Alternative 4 is chosen as the reference design for the purpose of sizing the power, weight, and volume of the C&DH subsystem. This alternative combines the qualities of Alternatives 2 and 3 to yield a switching technique with no mechanical RF switches and no reduced capability with the failure of a transmitter and/or receiver. In addition, a complete transponder can fail in this scheme and the system still retain the capability to transmit either FM or PM information over either antenna. This capability is available here since the transponders are identical.

D. System Description

The present C&DH System is summarized below. For further detail consult the Phase A LST Report.

1. Communications. The reference communications subsystem (Fig. X-6) consists of two unified S-band, modified ERTS transponders, both with FM and PM capability. For the purpose of commonality two modified ERTS transponders are now used in place of the one ERTS and one Apollo proposed in Phase A. This arrangement provides the engineering data a redundant PM data downlink for command verification, real-time engineering data, and tape recorded engineering and scientific data at a rate of 51.2 kb/sec. The scientific data have a redundant FM data downlink in either a digital or analog mode. The digital downlink rate for scientific information is 1 megabit/sec. This rate corresponds to the current single downlink capacity to ground stations for NRZ data streams.

This system uses four frequencies, two for the frequency diversified uplink and two for downlinks. Redundant information is sent from the ground to the LST on the two uplink frequencies. One of these frequencies is received through one of the antennas by the active receiver in the first transponder (the other receiver is in a standby condition) and sent to the PSK demodulator assembly in the data management equipment. The other uplink frequency is

received through the other antenna by the active receiver in the second transponder (one receiver is on standby) and also sent to the PSK demodulator assembly. This assembly uses the signal of highest strength and sends signals back to the communications equipment to switch the appropriate transmitter outputs to the antenna feeding the receiver with the highest automatic gain control (AGC) signal.

2. Data Handling. The design reference data handling configuration shown in Figure X-7 contains all the equipment required to manage the flow of data to and from the LST, SIP, and SSM. This includes the receipt, processing, and execution of real-time and stored commands; the processing, formatting, storage, and forwarding for transmission of all diagnostic and status information from all LST systems and subsystems, and the routing of scientific data for transmission to ground stations.

This data handling system is the one presented in the Phase A document. No major changes have occurred in this system since the publication of the Phase A Report. The only revision of significance pertains to the number and kind of measurements required within each LST subsystem. A revised measurement list is presented in Table X-1.

3. Hardware Summary. The C&DH System hardware summary is presented in Table X-2.

E. Projected C&DH System Impacts

Several ensuing developments could impact the present C&DH System design. Ground network improvements, Tracking and Data Relay Satellite System (TDRSS) development, image tubes improvements, and mass memory developments may influence the C&DH System design.

1. Ground Network Improvements. The Spacecraft Tracking and Data Network (STDN) is presently undergoing modifications that will increase its capability. One significant modification is the addition of a Multifunctional Receiver (MFR). Generally the MFR will receive frequencies from 100 MHz to 10 GHz and has selective bandwidth from 10 KHz to 30 MHz. Other modifications include capabilities for modified tone ranging and biased doppler techniques, updated computer systems for increased data handling capabilities, updated buffers for the command system, etc. All ground station modifications should be examined to determine their impacts upon the LST communication system.

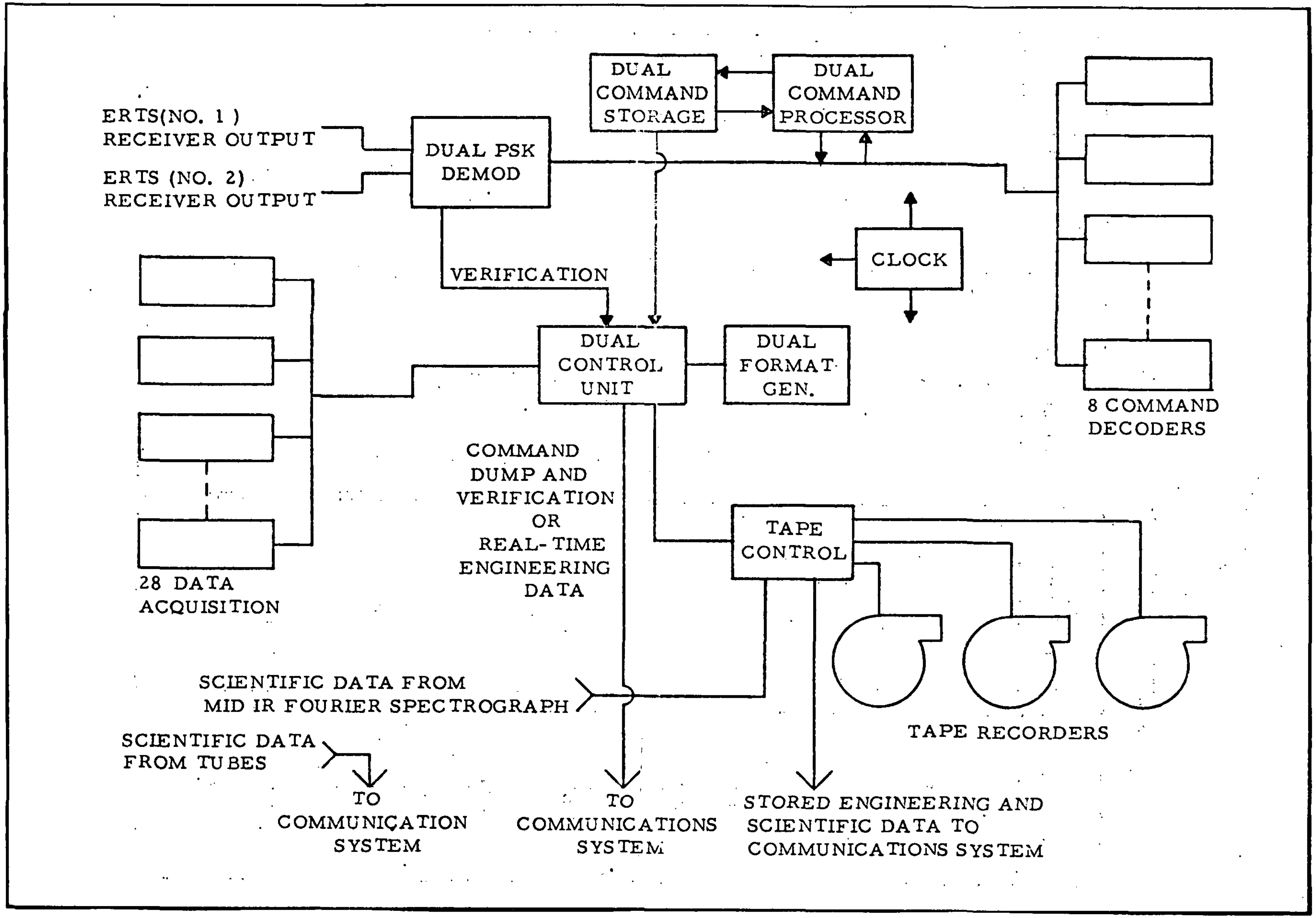


Figure X-7. Data handling system.

TABLE X-1. LST MEASUREMENT CHANNEL LIST

System	No. of Channels	Sample Rate Sample/Second	Total Number of Channels
ACS	15	10	
ACS	10 (2 word/Ch)	10	
			35
OTA	62	1	
SIP	5	1	
ACS	110	1	
			177
C&DH	100	0.2	
Power System	115	0.2	
ACS	142	0.2	
Structure	30	0.2	
			387
OTA	204	0.1	
SIP	29	0.1	
			233

TABLE X-2. COMMUNICATIONS AND DATA HANDLING HARDWARE SUMMARY

	No. of Units Required	No. of Units Provided	Unit Power (W)	Total Power (W)	Unit Heat Dissi-pated	Size		Unit Mass (Weight)		Total Mass (Weight)		Allowable Operating Temperature (°C)
						mm	in.	kg	lb.	kg	lb.	
Circulator Switch	2	2	3	6	3	101.6 x 101.6 x 25.4	4 x 4 x 1	0.9	2	1.8	4	-18 to 60
ERTS Transponder	2	2	28	9.3	9.3	203 x 152 x 330	8 x 6 x 13	10.9	24	21.8	48	-18 to 60
Antenna	2	2	--	--	--	Cone: 76D x 245H	3D x 10H	0.9	2	1.8	4	-18 to 60
PSK Demodulator & Logic	1	1	4	4	4	184 x 140 x 89	7.25 x 5.5 x 3.5	0.9	2	0.9	2	-18 to 60
Data Control Unit	1	2	3	3	3	152 x 229 x 71	6 x 9 x 2.8	1.8	4	3.6	8	-18 to 60
Format Generator	1	2	4	4	4	152 x 229 x 152	6 x 9 x 5.6	1.8	4	3.6	8	-18 to 60
Command Processor & Memory	1	1	16	16	16	107 x 130 x 366	4.2 x 5.1 x 14.4	6.2	13.7	12.4	27.4	-18 to 60
Data Acquisition Unit	14	28	1.34	1.34	1.34	71 x 71 x 25	2.8 x 2.8 x 1	0.1	0.25	3.2	7	-18 to 60
Command Decoders	8	16	1.4	11.2	11.2	152 x 229 x 71	6 x 9 x 2.8	0.4	0.8	6.4	12.8	-18 to 60
Tape Control	1	2	1	1	1	157 x 185 x 41	6.2 x 7.3 x 1.6	1.4	3	2.7	6	-18 to 60
Tape Recorder	3	3	12 rec 20 P.B.	20	20	246 x 203 x 147	9.7 x 8 x 5.8	5.5	12	16.4	36	-18 to 60
Clock	1	1	1.7	1.7	1.7	56D x 114H	2.2D x 4.5H	0.5	1	0.5	1	-18 to 60
Multicoupler	2	2	--	--	--	63.5 x 63.5 x 152.4	2.5 x 2.5 x 6	0.9	2	1.8	4	-18 to 60
Downlink Data Distributor	1	2	4	4	4	76.2 x 76.2 x 152.4	3 x 3 x 6	0.9	2	1.8	4	-18 to 60

2. <u>Tracking and Data Relay Satellite System Development</u>. The design reference communication subsystem for transmitting scientific data to earth does not allow a continuous look at high data rate experiments because of the relatively narrow information bandwidth of the STDN ground intercommunication system and the intermittent contact time of satellites in low to medium earth orbits. Use of the synchronous altitude TDRSS with its wide information bandwidth and nearly continuous contact time will allow a real-time look at high data rate experiments if the LST antenna can track the TDRSS without disturbing the pointing stability on the target. This real-time data transmission capability would allow the LST to utilize an image tube that is not constrained to have target storage capability. The Phase A design reference image tube was a secondary election conduction (SEC) vidicon that has the capability to store images on its target face, thus removing the requirement for an onboard mass memory. However, the SEC vidicon suffers from low resolution (approximately 20 cycles/mm). By removing the tube target storage constraint a higher resolution image tube could be utilized. For further information on TDRSS impacts upon the LST consult the LST Phase A document.

3. <u>Image Tube Improvements</u>. The impacts upon the C&DH System of utilizing an image tube with greater resolution than the SEC vidicon will depend upon the particular tube in question. Factors such as sensitivity, resolution, conductance, capacity, dark current, etc., will all have to be considered to determine the system impacts. Also included in these considerations should be the availability of the TDRSS and developments in the field of mass memories. If a tube with resolution significantly higher than the 20 cycles/mm of the SEC vidicon is required, it is possible that this tube will have a target with high conductivity (no target storage capability). Thus either the TDRSS will have to be available for real-time ground contact or a mass memory will have to be used that can store the image until ground contact can be established. For further information on candidate image tubes consult the LST Phase A Report.

4. <u>Mass Memories</u>. A memory large enough to store a single frame of data from an image tube required for the LST mission presents a significant technology problem. Several techniques were investigated prior to the Phase A report in hopes of finding a small, lightweight, low power, nonvolatile memory. No existing memory was found suitable for LST applications. The results of the memory survey are presented in the Phase A document. If the TDRSS is not available for LST use, developments in the mass memory field may be critical in determining whether or not a higher resolution image tube can be utilized.

F. Conclusions and Recommendations

The new guidelines concerning maintenance modes have no effect on the functional operation of the C&DH System.

Reducing redundancy in the C&DH area is not considered sound practice. Command transmission and verification, data transmittal, and status monitoring are vital necessities in the total LST mission success.

The requirement for antennas to be mounted on rollup solar arrays presents no major problems. Either solar panel configuration is compatible with the antenna subsystem. However, the FRUSA solar panel will allow better signal performance in the communications link. It is recommended that the patterns of the antennas be examined with the arrays in the stored position during Phase B to determine the amount of coverage available in this mode.

In the area of antenna switching four methods were proposed and analyzed. A switching technique utilizing circulator switches was chosen since this type switch is inherently more reliable than the mechanical RF switch and the switching arrangement will permit full system capability to be maintained with the failure of a transmitter and/or a receiver. In addition selective transmission of the scientific or engineering data is possible with the failure of a complete transponder. Coupled with this change in switching technique, the communications system now uses two modified ERTS transponders for reasons of commonality. The old system had one modified ERTS and one Apollo transponder.

A revised LST Measurement Channel List was presented for the Data Aquisition System.

The C&DH System design may be impacted by ground network improvements, the availability of TDRSS, and developments in image tube and mass memories. It is recommended that the improvements and developments in these areas be examined during Phase B to determine their possible influence upon the C&DH System.

CHAPTER XI. ATTITUDE CONTROL SYSTEM (ACS)

A. Guidelines and Constraints

Contractor Guidelines for the LST Project, which establishes an initial point of departure for the Phase B OTA/SI studies, [1] was searched for those guidelines which impact the updated SSM ACS. In general, the updated guidelines are more stringent, but less specific, than those used in the Phase A reference [2]. However, many of the more stringent items are given as goals or engineering targets subject to additional trades instead of hard ground rules. Examples are planet or comet viewing, spectrograph slit roll orientation, 0.025-arc second slit positioning relative to the entrance aperture, maneuvers programmed to avoid the sun, and a specified resolution of 0.03 arc second at 3350 Å. Additional interfaces between the OTA/SI and SSM ACS may be required for roll orientation, experiment slit positioning and planet or comet viewing which may require experiment related, but currently undefined, sensors. In addition, very low, but controlled body rates are required for planet tracking, scanning, and viewing. However, rapid slew rates may be required to prevent sunlight entering the OTA tube when inner planet viewing is terminated after having used the earth as an occulting disk.

For the Phase A design reference, 15 deg from the earth's limb or moon was specified as a constraint on observations, whereas the updated guidelines specify 5 deg from the earth's limb and 10 deg from the moon's center. The sun avoidance cone will depend upon the light shield truncation angle (45-deg half cone specified in Phase A). The system resolution of 0.03 arc seconds at 3350 Å is a deviation from the Phase A design reference which could impact overall pointing performance, stability criteria, and error budgeting between subsystems. For the Phase A design reference, a 0.05-arc second resolution at 5000 Å was used as a basis of establishing the 0.005-arc second overall motion stabilization requirement. At this time there appears to be no strong basis for tightening the pointing and stabilization requirements; hence the Phase A requirements are retained.

Based on the LST guidelines and assumed ground rules, a list of ACS design requirements was generated and summarized (see Table XI-1). Most of the requirements are self imposed and are subject to change, based on future trade studies. The nominal tipoff rate of 0.15 deg/s (not specified in Phase A) is a Shuttle specification that could be increased to the control moment gyro (CMG) capability to counteract LST motion (about 0.4 deg/s). However this CMG capability is not sufficient to compensate worst case Shuttle design requirements of 0.75 deg/s in all axes. To perform a 60-deg maneuver

TABLE XI-1. LST ACS REQUIREMENTS

1. Nominal tipoff rates will not exceed 0.15 deg/s.

2. Automatic sun acquisition, or reacquisition after a failure or loss of reference.

3. The momentum exchange system will be sized to store worst case momentum accummulation over a one-orbit interval.

4. The momentum exchange system will be sized for a peak turning rate of at least 3 deg/min.

5. The SSM ACS acquisition accuracy will be 30 arc sec for two axes and 0.1 deg about the LOS to bring guide stars within the OTA FGS FOV.

6. Using attitude information from the FGS, the SSM ACS will provide LST body pointing of 1 arc second.

7. The OTA FGS will provide LOS experiment pointing of 0.1 arc second and stabilization of 0.005 arc second.

8. The MTS will be sized to counteract maximum environmental torques.

9. An RCS will be sized and used for emergencies, abnormal control situations, and backup to 2 weeks (total impulse 3200 lb-sec).

10. Prior to LST release from the Shuttle the CMGs will be spun up and the SSM ACS will be activated.

11. The light shield truncation angle is 45 deg.

in 40 min would probably require peak turning rates of twice the average rate; hence a 3-deg/min peak is assumed. A Magnetic Torquer System (MTS) sized only to dump the momentum accumulation caused by environmental torques has only about one-half the capacity of one sized to counteract the maximum torque. The maximum torque sizing criteria assure operation even with partial MTS failure. The pointing specifications are assumed to be unchanged from the Phase A reference. The 30-arc second acquisition accuracy is based on trade studies between the expected best performance of body-mounted star trackers and the acquisition field of view (FOV) of the OTA fine guidance sensor (FGS). Assuming FGS attitude information, the SSM actuators are required to provide 1-arc second body pointing at which time OTA FGS can reduce its FOV and the secondary mirror control system can provide 0.1-arc second pointing and 0.005-arc second stability. Currently, there are indications that the SSM actuators could be selected to satisfy the requirements of 0.1-arc second body pointing and 0.005-arc second stabilization if an appropriate experiment related attitude error signal were provided. This would permit elimination of the secondary mirror control system, but would probably require additional small reaction wheels or small CMGs to augment the larger CMG system. The OTA FGS would be retained and augmented with additional experiment related sensors as specified during the Phase B study.

Prior to LST release from the Shuttle, the CMGs will be spun up and the ACS activated. Maximum use will be made of the MTS to damp tipoff rates, for momentum dump, to provide direct control torques, and if needed, to avoid CMG gyro hangup. In normal situations, the reaction control system (RCS) will never be used. However, the RCS is needed to satisfy the autonomous attitude hold guideline and is sized for emergencies and backup control modes for approximately 2 weeks. As an assumption, on-orbit, unpressurized maintenance of the RCS will be performed by extra-vehicular activity (EVA). With the exception of the change in the maintenance mode, the RCS guidelines and assumptions are essentially the same as those listed in the Phase A report [2] and Reference 3.

The 2 1/2-year period before the first ground-return maintenance does not permit a reduction in either reliability or redundancy. Moreover, there has been no reduction in the required ACS functions or operational modes that would permit a reduction in the ACS equipment or subsystems. Hence, there is a minimum impact on the ACS in going from an on-orbit pressurized to a ground-return maintenance mode. There could be a reduction in Shuttle payload support equipment, but in general the maintenance mode is not an SSM ACS configuration driver.

B. Impulse Requirements

1. Typical LST Characteristics. Typical vehicle mass, inertia and maximum gravity gradient (gg) torque characteristics are shown in Table XI-2 for four LST configurations. The configuration 1A represents a short SSM shell designed for earth-return maintenance, and configuration 2B represents a minimum deviation from the Phase A reference designed for minor on-orbit maintenance. Both configurations are also shown with a 20-percent increase in distributed mass. The mass varies from about 4536 kg (16 000 lbm) to 9072 kg (20 000 lbm), and the inertia values vary from about 16 270 kg-m^2 (12 000 slug ft^2) on the minor axis of inertia to about 98 975 kg-m^2 (73 000 slug-ft^2) on the major. Although the updated LST SSM is shorter than the Phase A design reference, the subsystems are grouped more toward the aft end, and the OTA/SI is assumed to be unchanged. Consequently, the updated inertia values are similar to the Phase A design reference, and the environmental torque and resulting momentum requirements are relatively unchanged. As listed in Table XI-2, the maximum gg torque is 0.142 N-m (0.105 ft-lb) and the maximum momentum required to obtain a peak turning rate of 3 deg/min is 87 N-m-s (64 ft-lb-s). Both values apply to configuration 2B with a 20-percent mass increase.

2. Momentum storage requirements. Momentum requirements for the 1A and 2B SSM configurations are shown in Table XI-3. The cyclic momentum represents the worst case gg torque over a one-quarter orbit interval. If the gg torque is biased, then a slow buildup in momentum occurs. The secular momentum represents the maximum buildup caused by gg torque over a one-orbit period. Based on the requirement that the momentum exchange system will be sized to store worst case momentum accumulation over a one-orbit interval, the secular momentum plus 10 percent added for other environmental torques is used as the requirement for sizing the CMGs. For all cases, the momentum needed to attain a vehicle rate of 90 deg in 5 min is shown to be only slightly higher than this requirement. The last column of Table XI-3 represents the minimum momentum storage required for attitude control. Assuming that the MTS continuously dumps the secular momentum, the control minimum is the sum of the cyclic plus the 3-deg/min maneuver momentum requirements. In the event of a CMG failure, the remaining CMGs must be capable of providing the control minimum to meet the LST attitude control requirements without degrading performance. Since the long SSM configuration is typical of the LST configuration analyzed, its momentum requirement of 452 N-m-s (333 ft-lb-s) is used as a basis for sizing the CMGs. After one CMG failure, the remaining CMGs must provide 212 N-m-s (156-ft-lb-s) momentum to meet the control minimum criteria.

TABLE XI-2. UPDATED LST CHARACTERISTICS

Configuration	Axis	Ctr of Mass m (ft)	RCS Lever m (ft)	Inertia kg-m^2 (slug-ft^2)	Max gg Torque N-m (ft-lb)	3-deg/min Momentum N-m-s (ft-lb-s)
Earth-return maint	X	3.75	1.83	16 338	0.0087	14.3
(Conf 1A)		(12.31)	(6)	(12 050)	(0.0064)	(10.53)
7294 kg (16 080 lb)	Y	0.0179	3.66	77 306	0.1151	67.5
		(0.0588)	(12)	(57 018)	(0.0849)	(49.75)
	Z	-0.0025	3.66	82 289	0.1064	71.7
		(-0.0081)	(12)	(60 693)	(0.0785)	(52.92)
Minor on-orbit maint [a]	X	4.42	1.83	16 687	0.0087	14.6
(Conf 2B mod)		(14.49)	(6)	(12 308)	(0.0064)	(10.74)
7384 kg (16 279 lb)	Y	-0.0179	4.42	92 483	0.1410	80.7
		(-0.0587)	(14.5)	(68 212)	(0.1040)	(59.52)
	Z	-0.0279	4.42	97 471	0.1323	85.1
		(-0.0916)	(14.5)	(71 891)	(0.0976)	(62.73)
1A +20 percent	X	3.93	1.83	17 429	0.0087	15.2
8753 kg (19 296 lb)		(12.91)	(6)	(12 855)	(0.0064)	(11.22)
	Y	0.1494	3.96	79 349	0.1167	69.2
		(0.4900)	(13)	(58 525)	(0.861)	(51.07)
	Z	-0.0020	3.96	84 335	0.1081	73.6
		(-0.0067)	(13)	(62 201)	(0.0797)	(54.28)
2B +20 percent	X	4.578	1.83	17 794	0.0087	15.5
8861 kg (19 535 lb)		(15.02)	(6)	(13 124)	(0.0064)	(11.45)
	Y	-0.0149	4.57	94 225	0.1421	82.2
		(-0.0489)	(15)	(69 497)	(0.1048)	(60.64)
	Z	(-0.0233)	4.57	99 213	0.1334	86.6
		(-0.0763)	(15)	(73 176)	(0.0984)	(63.85)

a. Configuration reference used for momentum requirements.

TABLE XI-3. LST MOMENTUM REQUIREMENTS

Configuration	Axis	Momentum N-m-s (ft-lb-s)				
		Cyclic [a]	Secular [a]	90 deg/5 min [b]	Req'd [c]	Control Minimum [d]
Earth-return maint (Conf 1A)	X	8.1 (6.0)	25 (18.7)	86 (63.1)	28 (20.5)	22 (16.5)
	Y	106.7 (78.7)	335 (247.3)	405 (298.5)	369 (272.0)	174 (128.4)
	Z	98.7 (72.8)	310 (228.6)	431 (317.8)	341 (251.5)	170 (125.7)
Minor on-orbit maint (conf 2B mod)	X	8.1 (6.0)	25 (18.7)	87 (64.4)	28 (20.6)	23 (16.8)
	Y	130.7 (96.4)	411 (302.9)	484 (357.1)	452 (333.2)	211 (155.9)
	Z	122.7 (90.5)	385 (284.2)	510 (376.4)	424 (312.6)	208 (153.2)

a. Based on maximum gg torque (one-fourth orbit cyclic, 1 orbit secular).
b. 90 deg/5 min = 0.005, 2356 rad/s is a design goal, but is not a requirement.
c. The secular + 10 percent for other environmental effects.
d. MTS dumps secular. The minimum for control is the sum of cyclic plus 3 deg/min requirement.

3. <u>CMG Sizing</u>. Assuming the four skewed single gimbal CMG (SGCMG) configuration with a skew angle of 30 deg (same as Phase A design reference), the momentum requirement per CMG, total in the CMG system, and the amount per vehicle axis are shown in Table XI-4 for several criteria. Assuming one CMG is inoperable and the remaining three CMGs provide momentum, 81 N-m-s (60 ft-lb-s) per CMG is required to meet the control minimum criteria, 176 N-m-s (130 ft-lb-s) to meet the full requirement and 292 N-m-s (215 ft-lb-s) to meet the full requirement with a 40-percent reserve. With all four CMGs operating, 122 N-m-s (90 ft-lb-s) per CMG meets the momentum requirement for the LST. Since there is considerable question about the ability of SGCMGs to fully utilize all their capability without encountering either a control singularity or an internal gyro hangup condition, the CMGs could be oversized to operate in a region of momentum space that is free of singularities and hangups. Based on studies in which the initial gimbal angle sets were varied and the singularity free momentum space calculated, about 60 percent of the total momentum envelope centered about the origin can be used without encountering singularities. For this reason, 203 N-m-s (150 ft-lb-s) CMG units are recommended for the LST to permit one-orbit operation under worst case environmental torques without dump in a singularity free momentum region. This CMG size is large enough to allow reasonable vehicle growth without resizing, and meet the one-orbit requirements with any one CMG inoperative. With four CMGs, a peak rate of 26.8 deg/min is obtainable, hence a 90-deg in 5 min maneuver should be practical. Three CMGs operating can produce a peak rate of 18.6 deg/min which will probably not be adequate for a 90-deg in 5 min maneuver, but the normal maneuver requirement is more than satisfied. This CMG size also permits the expected Shuttle release and misdocking rates to be counteracted without using the RCS.

The same sizing criteria were applied to four skewed reaction wheels (RWs). It was concluded that RWs must be sized at 237 N-m-s (175 ft-lb-s) per unit to meet the momentum requirement, in which case the control minimum requirement is also met with any one RW inoperative. Reaction wheels can satisfy the 60-deg/40-min maneuver requirement, but the goal of 90 deg/5 min cannot be satisfied without a severe power penalty (about 1000 W). Large RWs do not provide the growth potential or the high torque output of CMGs; therefore a pure RW system is not recommended for LST.

Although SGCMG of the size recommended for the LST is not an off-the-shelf unit, the estimated mass of a 203 N-m-s (150 ft-lb-s) SGCMG unit is 45.4 kg (100 lbm), plus 9 kg (20 lbm) if the drive electronics are separated from the CMG. Estimated power requirements are 85 W peak during a 4-hour spin up period, 10 W during normal pointing, and 17 W during maximum maneuvers. A smaller CMG unit could easily be used to meet the LST requirements by simply using more units in a skewed configuration. Smaller CMG units would probably be more applicable to a class of smaller vehicles,

TABLE XI-4. LST CMG SIZING CRITERIA

CMG Sizing Criteria	Momentum N-m-s (ft-lb-s)[a]				
	Per CMG	System	X	Y	Z
3 CMG cont. min[b]	81 (60)	244 (180)	122 (90)	212 (156)	222 (164)
3 CMG req'd[c]	176 (130)	529 (390)	264 (195)	458 (338)	481 (355)
3 CMG 60-percent use[d]	292 (215)	875 (645)	438 (323)	758 (559)	796 (587)
4 CMG cont. min	61 (45)	244 (180)	122 (90)	228 (168)	228 (168)
4 CMG req'd	122 (90)	488 (360)	244 (180)	456 (336)	456 (336)
4 CMG 60 percent use	203 (150)	813 (600)	407 (300)	759 (560)	759 (560)

a. Assuming a 30-deg skew angle for four skewed CMGs.
b. Assuming CMG no. 4 is out.
c. Assuming all the momentum is usable.
d. Assuming 60 percent of the momentum is usable.

such as the new HEAO or the proposed EOS. Moreover, recent contractor (Martin) studies have developed singularity and hangup avoidance steering laws for SGCMG clusters of more than four CMGs. A 6-skewed configuration using 136-N-m-s (100-ft-lb-s) CMG units is an attractive alternate for the LST.

4. Magnetic Torque System Sizing. Since the torque and momentum sizing requirements for the updated LST are similar to the Phase A design reference, no change is recommended in the electromagnet (EM) sizing. However, based on contacts with Ithaco, Inc., curved EMs pose manufacturing problems. Ithaco proposed the use of bent EMs instead of the curved EMs in the Phase A design reference. Studies indicate that bent EMs can easily be installed in the OTA without either a power, weight or size penalty. The basic function of the MTS is to provide momentum management of the CMGs. As secondary functions, the MTS provides direct control torque, if needed, and can be utilized to prevent gyro hangup and possibly provide smoother CMG operation.

5. RCS Sizing. The RCS is included on the LST as a backup control system. Therefore, only control situations that may be considered of an emergency nature will be used in sizing the RCS. Two thrust levels are considered desirable. Situations involving docking, Shuttle release, or large uncontrolled CMG torques could require up to 44.5-N (10-lb) force to produce sufficient control authority. During a failure of the primary control system, however, normal sun pointing can be accomplished more efficiently with a small thrust level, about 2.2 N (0.5 lb).

Table XI-5 illustrates a typical impulse budget that was assumed for the RCS. During despin of the CMGs after an ACS failure, the RCS must counteract internal and external torques. For this case, the impulse required is about 623 N-s (140 lb-s). To maintain power during an emergency, the solar panels must be oriented toward the sun. Using only two-axis control (allowing roll about the sun line) the RCS should be able to perform for about two weeks at a cost of about 11 121 N-s (2550 lb-s). For Shuttle retrieval, the LST may be required to attitude hold one orbit during inspection, preliminary docking procedures, etc., requiring about 67 N-s (15 lb-s). Misdock was assumed to impart rates equal to the worst case design requirements for release transients (0.75 deg/s on all axes). With that assumption and assuming three attempts before a hard dock, the allocated impulse budget is 2046 N-s (460 lb-s). Docking (including attitude hold for further inspection) requires 200 N-s (45 lb-s). The total impulse budget for RCS is 14 279 N-s (3210-s) which should meet the requirements within the assumptions made. It should be noted, however, that the allocated impulse budget when compared to actual usage can vary widely as different emergency situations are considered.

TABLE XI-5. LST RCS IMPULSE BUDGET (LONG SSM)

Event	Impulse	
	N-s	(lb-s)
Attitude hold during CMG despin (± 0.5 deg)	623	(140)
Attitude hold for 2 weeks (± 0.5 deg)	11 343	(2550)
Attitude hold for 1 orbit for Shuttle docking (± 0.5 deg)	67	(15)
Misdock (0.75 deg/sec in 3 axes) 3 attempts before hard dock	2 046	(460)
Docking	200	(45)
Total	14 279	(3210)

NOTE: Assuming: Dual level RCS (10 lb/0.5 lb); specific impulse of 65

C. ACS HARDWARE SUMMARY

A functional block diagram of the LST ACS is illustrated in Figure XI-1, including key interfaces with the OTA/SI system components. The ACS is shown with a complement of sensors, actuators and interface equipment required to perform all phases of the LST mission. Indirect interfaces between the ACS components and the LST structure are indicated by dashed lines, such as magnetic field propagation, CMG vibration and control system interaction with structural modes. Direct interfaces between the ACS and secondary mirror control system and FGS are based on the Phase A design reference, and are the interfaces most likely to be altered as a result of the OTA/SI Phase B study. Utilization of the ACS components to perform the required mission modes are fully described in the Phase A LST report [2]. An updated hardware summary for the SSM ACS is provided in Table XI-6, which is identical to the Phase A design reference except for the CMGs, coarse sun sensors (CSSs) and RCS.

Only two CSSs are used (five in Phase A) to provide 2 π steradians for sun acquisition or reacquisition after loss of attitude reference. The three CSSs which have been deleted are the one on the antisolar side of the SSM and the two on the solar wings. The CSS on the antisolar side of the SSM served basically as a solar indicator to establish a bias command that causes the positive Z-axis to be rotated into the sunlight. This type of command with relatively simple search logic can just as easily be established by the absence of the sun for either of the two CSSs on the sunward side of the LST.

Two three-axis magnetometers are provided to sense the earth's magnetic field and generate the vector components of the field in spacecraft body coordinates. Since the earth's field changes slowly as the spacecraft orbits the earth, the magnetometer outputs can also be processed to generate coarse rate information for initial stabilization. The CSS and the magnetometer can provide the signals required to null body rates, for sun acquisition, and to hold a coarse pointing mode with the position about the sunline uncontrolled. Since the six EMs listed as magnetic torquers in Table XI-6 are located forward on the OTA, the magnetometers are mounted as far aft on the SSM as possible to minimize the interference between the EMs and magnetometers. Only one of the three-axis magnetometers is normally used, with the other being redundant.

Three fixed star trackers (FSTs) are mounted on the primary mirror support structure and view out the antisolar side of the SSM. One FST is aligned with the negative Z-axis and the other two are skewed 45 deg in the Y-Z plane relative to the first one. Normally, only two FSTs are active with

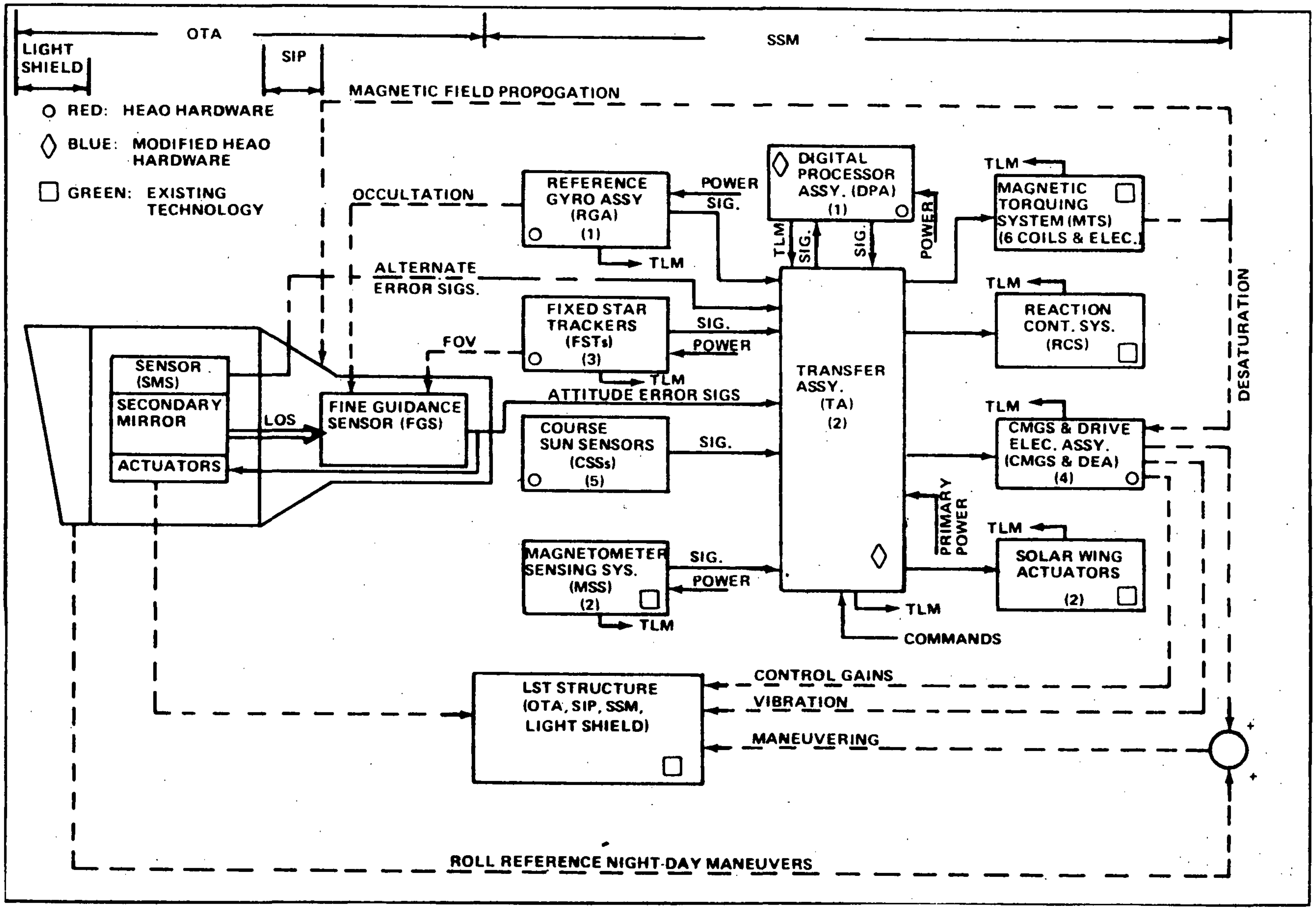


Figure XI-1. ACS block diagram and key interfaces for LST.

TABLE XI-6. LST ACS HARDWARE SUMMARY

Item	Ident No.	Ident Mod.	Number of Units	Power (W) Unit	Power (W) Ave	Mass (kg) Unit	Mass (kg) Total	Total Volume (L)	Unit Size (mm)
Coarse sun sensor	7	X	2	0.0	0.0	0.1	0.2	0.14	41 x 47 D
Magnetometer	5	X	2	2.0	2.0	3.0	6.0	8.20	254 x 127 x 127
Fixed star tracker	3		3	5.0	10.0	5.5	16.5	15.83	406 x 114 x 114
Star tracker shade	3		3	0.0	0.0	1.6	4.8	100.11	457 x 305 D
Ref gyro assembly [a]	4		(6 gyros)	(9/gyro)	36.0	10.4	10.4	11.68	444 x 173 x 152
Transfer assembly	12	E	2	26.5	26.5	4.7	9.4	15.18	246 x 203 x 152
Digital processer assy	11	E	1	16.0	16.0	6.2	6.2	5.09	366 x 130 x 107
Magnetic torquer [b]	2	OTA	6	5.0	(30peak)	23.5	141.0	23.34	1905 x 51 D
Magnetic torquer elect	6	E	1	5.0	5.0	2.3	2.3	2.11	203 x 102 x 102
SGCMG [c]	1	D	4	10(17)	40(68)	45.4	181.6	672.80	813 x 488 x 424
RCS (GN_2) [d]	15	I	1	peak	5.0	123.0	123.0	(Est. Sys)	660 D-Tank
RCS elect	13	-	1	11.0	0.0	7.3	7.3	14.51	356 x 229 x 178
System totals					155.5		508.7	1041.79	

a. Four gyros operational

b. Peak power of 30 W

c. Power during slew shown in ( ).

d. Peak power of 131 W when RCS use is initiated.

one being redundant. After the LST has been rate stabilized and the sun acquired, the FSTs are used to obtain an inertial reference and provide three-axis attitude error signals. The FST must be accurate enough to ensure that preselected guide stars appear within the coarse FOV of the FGS on the OTA when the LST is in a coarse celestial pointing mode. Based on Phase A trade studies, the LST coarse pointing accuracy is 30 arc seconds with 0.1 deg roll about the LOS. After OTA guide star acquisition, three-axis attitude error signals are obtained from the OTA FGS, instead of the FST, and are used to reduce the body-pointing error to 1 arc second for each axis. In case of OTA FGS occultation, the FSTs are used to update the Reference Gyro Assembly (RGA), maintaining coarse body pointing during the period.

The Reference Gyro Assembly (RGA) consists of six gyros arranged in a skewed dodecahedron configuration with the necessary support electronics. During normal operation, four gyros are active and two are in a redundant standby mode. Since the RGA must be aligned with the FSTs, it is located in the same area of the spacecraft and on the same structure to provide an accurate operational interface. During spacecraft fine pointing operation, the RGA must be accurate enough to provide 1 arc second body pointing with frequent updates from the OTA FGS. During OTA FGS occultation, the drift of the RGA must be small enough to hold the spacecraft within the coarse pointing accuracy of 30 arc seconds with updates from the SSM FSTs. The RGA is normally used during all LST operational modes, as well as autonomous attitude hold modes.

The two transfer assemblies (TAs) serve as an interface between all ACS components. The input/output section provides an interface between the sensors, DPA and actuators, including multiplexing, D/A and A/D conversions, and signal conditioning. A power switching and converter section routes power at the required voltage levels to the ACS components and provides isolation. The command and telemetry section provides data buffering, command storage, and processing functions. The sensor buffer unit places the required sensors on line for the control mode in use and routes signals between sensors and the DPA. Hardwired functions as required by the autonomous attitude hold guideline are also contained in the TA. Normally one TA is active with the other unit on standby.

The digital processor assembly (DPA), which is internally redundant, receives input data and provides processed output data by way of the TA. The design reference DPA is a version of the Control Data Corporation 469 that was also selected for HEAO. Six 2000-word memory modules are used with 16-bit instruction and data words. The memory can be expanded to 64 000 words

in 2000-word increments. Typical processing and output data provided by the DPA are CMG gimbal commands, MTS commands, gyro drift compensation and solar wing actuator commands.

The magnetic torquer consists of six bar electromagnets (EMs) located on the forward end of the OTA. The physical size, location and characteristics of the EMs are the same as the Phase A design reference. The MTS consists of the magnetometer, Magnetic Torquer Electronics (MTE) and EMs. Cross product dipole commands are generated by the DPA, and the MTE converts these commands into torquing currents which drive the EMs. Consideration should be given to hard wiring the MTS for emergency or backup modes, and extending its use for coarse rate stabilization.

Four SGCMGs, each with a redundant drive electronics assembly (DEA), are mounted in a skewed configuration about the OTA LOS axis. The recommended skew angle is 30 deg to provide a momentum envelope somewhat proportional to the LST inertia ellipsoid. The configuration is the same as the Phase A design reference, but the size of each CMG unit is different. The Phase A CMGs were selected for HEAO commonality. However, the HEAO has been redesigned to a much smaller configuration and CMGs are not currently under consideration for it. Therefore, the LST CMGs have been resized specifically for the updated vehicle inertias and expected environmental torques. The result is a smaller CMG unit requiring less weight and power than the Phase A design reference. Since the ACS is activated before Shuttle release and the CMGs are large enough to satisfy all normal impulse and control requirements, the CMGs provide control torques for all normal LST modes, including nulling tipoff rates and attitude hold during docking. Commands that drive the CMG gimbals are originated within the DPA and routed through the TA to the drive electronics.

The RCS serves as a backup control system that is not used in normal ACS control modes or situations that the CMGs can handle. The RCS is placed in a standby go condition during critical LST control situations, such as Shuttle release of the LST and during docking or misdock recovery of the LST and Shuttle. If there is a partial or complete failure of the primary ACS, the TA would provide hardwired logic from the CSS, and magnetometer or RGA to permit the RCS to control the LST while the ACS is being checked out or until Shuttle retrieval.

The RCS described herein represents the Phase A design reference change necessary to reflect ground-return maintenance as well as on-orbit, unpressurized maintenance. The Phase A design reference RCS is described in detail in Reference 3. The two RCS maintenance concepts are illustrated in

Figure XI-2. The on-orbit, unpressurized maintenance concept utilizes quick disconnects and flex hoses for ease in replacement of various assemblies, whereas the ground maintenance concept utilizes hard fittings and brazed connections. Operationally, both RCS concepts are identical.

A pressure-regulated, gaseous nitrogen (GN_2) propulsion system has been retained because the impulse budget is low enough that the resulting total mass of the RCS is not critical, and GN_2 is not considered to be a contamination producing source. The RCS is modularized into four basic assemblies: a propellant tank, a regulator, and two thruster modules. These assemblies are outlined in phantom as shown in Figure XI-2 and represent the replaceable elements during maintenance. Auxiliary items associated with the RCS are latching solenoid isolation valves, pressure and temperature transducers, pressure gauges, manual shutoff valves, filters, pneumatic disconnects, propellant fill and drain valve, flex hoses, interconnecting plumbing, wire harness, and electrical connectors. A mass statement for the two selected RCS concepts is presented in Table XI-7.

Based on the typical impulse assumed in Table XI-5, a larger tank than that used in References 2 and 3 was selected. The new tank is an existing component used in the Skylab Thrust Attitude Control System (TACS). The tank is spherical with a diameter of approximately 0.61 m (2 ft), weighs 53.07 kg (117 lbm) empty, and is man rated. In a fully loaded operating condition, the tank contains 28.58 kg (63 lbm) of GN_2 at a pressure of approximately 2.14×10^7 N/m^2 (3100 psi). This propellant loading results in a nominal total impulse available of 16 378 N-s (3 682 lb-s), which is 2 100 N-s (472 lb-s) above the impulse budget shown in Table XI-5. An assumed specific impulse of 65 seconds was used to determine a propellant requirement of 22.7 kg (50 lbm). For the on-orbit, unpressurized maintenance concept, the tank will need to be modified to include a protective cover, a handle for ease in maneuvering, and structural supports for ease in loading the tank into its appropriate position on the LST.

The dual level pressure regulator and thruster modules selected in Reference 2 or 3 are still retained, except the backup regulator and thruster modules have been deleted. Only two thruster modules are used on the updated LST, with a total of six thrusters (three thrusters per module). With the regulator operating in the high mode, the thrust level is 44.48 N (10 lbf). Whereas with the regulator operating in the low mode, the thrust level is 2.22 N (0.5 lbf) through the same thrusters. The high and low regulated pressures are 6.89×10^5 N/m^2 gauge (100 psig) and 3.45×10^4 N/m^2 gauge (5 psig), respectively. Both the thruster modules and the pressure regulator are existing components used in the Agena RCS.

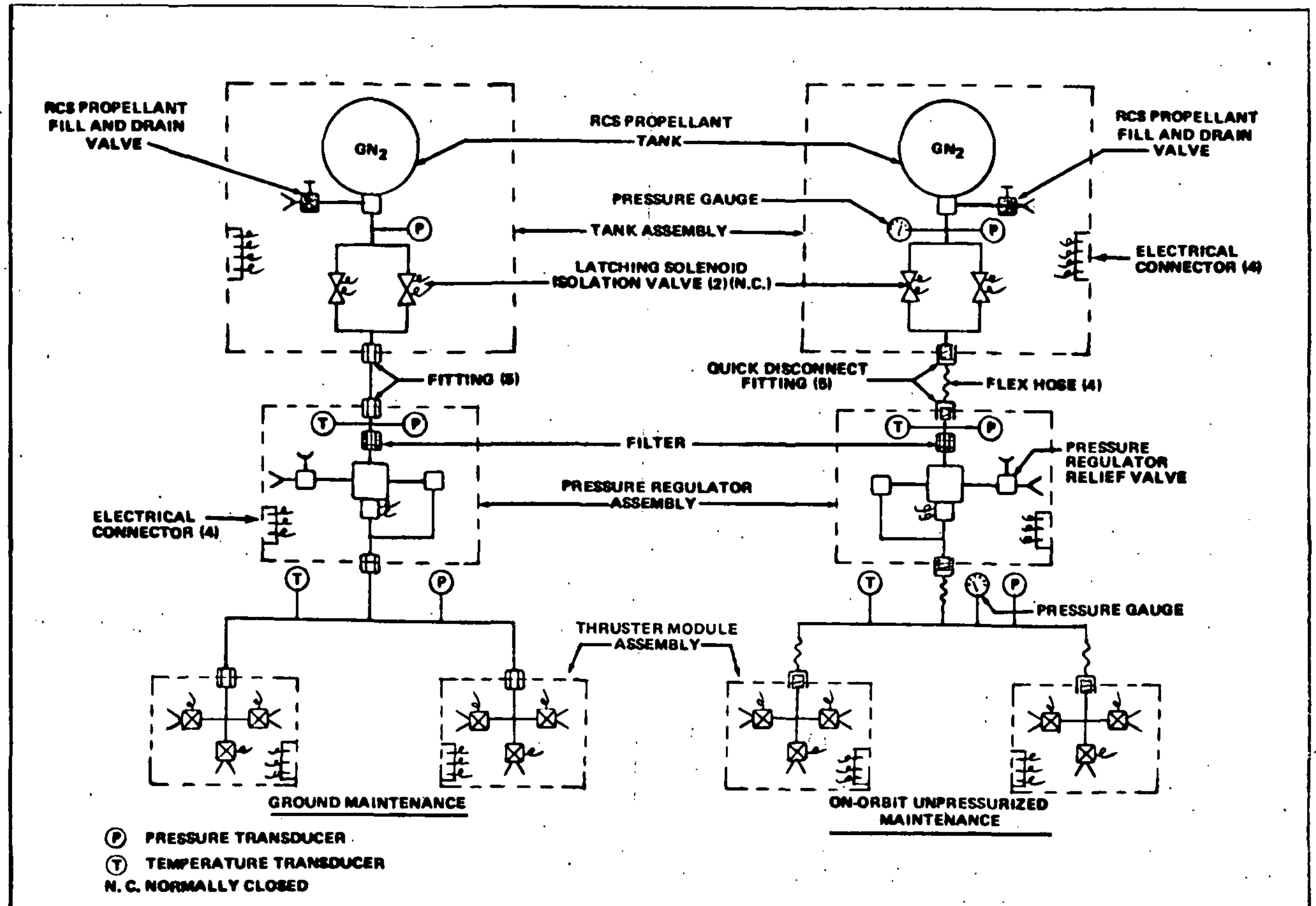


Figure XI-2. Delta Phase A LST reaction control system design.

TABLE XI-7. REACTION CONTROL SYSTEM MASS SUMMARY

Item	Mass ~ kg (lbm)			
	Ground Maintenance		On-Orbit Unpressurized Maintenance	
Tank	53.1	(117.0)	53.1	(117.0)
Fill and drain valve	0.5	(1.0)	0.5	(1.0)
Latching solenoid isolation valve (2)	3.6	(8.0)	3.6	(8.0)
Pressure regulator	3.6	(8.0)	3.6	(8.0)
Pressure transducer (3)	0.4	(0.9)	0.4	(0.9)
Temperature transducer (2)	0.2	(0.5)	0.2	(0.5)
Pressure gauge (2)	—	—	0.4	(0.8)
Thruster module (2)	4.5	(10.0)	4.5	(10.0)
Plumbing and fittings	4.5	(10.0)	5.4	(12.0)
Retention and miscellaneous structure	13.6	(30.0)	22.7	(50.0)
System dry mass	84.0	(185.4)	94.4	(208.2)
Propellant (GN_2) [a]	28.6	(63.0)	28.6	(63.0)
Total system mass	112.6	(248.4)	123.0	(271.2)

a. Redundant propellant:
Ground Maintenance: ≈ 5.9 kg (≈ 13.0 lbm)
On-Orbit Unpressurized Maintenance: ≈ 5.9 kg (≈ 13.0 lbm)

NOTE: Required total impulse = 14 278 N-s (3210 lb-s); total impulse available (nominal) = 16 378 N-s (3682 lb-s)

Since the RCS normally will never be used, the GN_2 will be confined to the tank assembly by a parallel arrangement of two closed latching solenoid isolation valves to minimize leakage and prolong the life of the regulator and thrusters. The life of the regulator and thrusters in this case will only be constrained by shelf life. A parallel arrangement of the two latching solenoid isolation valves is used to provide redundancy in opening, and is the only hardware redundancy in the subsystem. The RCS has now been designated entirely as a backup subsystem; hence, the redundancy shown in the Phase A design reference has been deleted.

The average electrical power required for components in the RCS is summarized in Table XI-8. The latching solenoid isolation valve requires 126 W for initial operation, after which the power requirement decreases. When not in use, only 5 W is needed for the RCS transducers to monitor continuously the status of the subsystem.

TABLE XI-8. REACTION CONTROL SYSTEM POWER REQUIREMENTS.

Item	Average Power (Watts)
Latching solenoid isolation valve	126
Pressure regulator	42
Thruster	18
Pressure transducer (total)	3
Temperature transducer (total)	2

NOTE: When not in use, the RCS requires approximately 5 W of continuous electrical power.

Based on a summary of the ACS hardware Table XI-6, the total ACS average power usage is 155.5 W, the weight is 508.7 kg (1121 lb) and the volume is 1042 liters. Compared with the Phase A design reference, the average power has been reduced by 23 W and the weight by about 130 kg (288 lb) most of which is due to smaller CMGs. However, the RCS weight increased by about 20 kg (44 lb) because of a larger tank containing more propellant.

As previously stated, the maintenance mode is not an SSM ACS configuration driver. However, if the decision were made to go to an extensive on-orbit EVA, manipulator, or robot maintenance mode, equipment grouping and packaging of components into modules would probably be required. Alignment problems for sensitive components would be compounded, and modular structural compliance could cause vibration propagation and closed-loop feedback problems. Criteria for grouping components and rationale for packaging into modules require further study during Phase B. New techniques and designs are required for module probes, latches, plugs, alignment and checkout. Shuttle support equipment would probably increase with the degree of modularity, as would the basic SSM weight.

There is no physical reason why most of the ACS components could not be modularized, with the exception of the FST and RGA. The FST must be aligned with the OTA fine guidance sensors to within 1.0 arc second and hence should be hard mounted to the OTA structure in the focal plane region. The RGA must be aligned to the FST, but some misalignment could be corrected by the DPA. Based on current requirements, both the FST and RGA should be aligned accurately to stable structure as recommended in the Phase A report. The electrical portions of the ACS, such as the DPA, TA, MTE, and RCS electronics, are amenable to modularization and all could be placed in one module. The RCS thrusters are already clustered into two modules on the aft end of the SSM. Both the sun sensors and the magnetometers must be exposed to the external spacecraft environment, with the sun sensors viewing out the positive Z-axis (sunward) side of the SSM and the magnetometers located as far aft as possible on the SSM to minimize the field effects of the EMs, which are located as far forward on the OTA as possible. Both of these sensors could be placed in one module located on the aft sunward side of the SSM. Each CMG or RCS tank is large enough to fill a single module, whose location could be almost anywhere. The CMG electrical portions are less reliable than the mechanical assemblies, and pending further studies, could be separated from the basic CMG. The electrical drive assemblies could be modularized for easy replacement with each CMG hardmounted to the SSM structure. As previously stated, the EMs are mounted on the forward end of the OTA embedded within the structure and hence are not easily modularized. Presently, each ACS subsystem is partially modularized; for example, each has its electronics in separate boxes. The type of modularization discussed above is simply arranging small modules into groups (packaging) to form larger modules, which appears to be a very inefficient method. Additional thought should be given to basic subsystems designed for modularization into standard trays to minimize interfaces between subsystems. For example, a standard motherboard could be designed to accept plug-in electronic modules, each having common connections, input power, and standard input/output formats.

D. CONCLUSIONS AND RECOMMENDATIONS

The LST maintenance mode selection has very little impact on the ACS configuration selection. The ground-return LST configuration has mass and inertia characteristics similar to the Phase A reference. Consequently, the environmental torques and actuator sizing criteria are also similar to those of the Phase A reference. Based on the new LST contractor guidelines [1], the attitude control pointing and stability requirements are unchanged. Hence, the Phase A selection of sensors, electronics and actuators [2] is still valid for the updated LST SSM ACS. However, some of the Phase A ACS component selections were based on a commonality with HEAO guideline, especially the CMGs. Since CMGs are currently not under consideration as a candidate actuator for the new HEAO, the LST CMGs have been resized downward, based on LST requirements and not commonality with other programs. A skewed SGCMG configuration using the pseudo inverse steering law is still recommended for LST [4]. However, alternate CMG configurations, such as the recently developed OMEGA [5], more than four SGCMGs in a skewed configuration [6], or DGCMGs are still viable candidates for which singularity avoidance steering laws have been developed.

The CMG vibration through the SSM structure is still an area of concern, but several recent studies [7] have shown that shock mounts can be used to keep the induced motion within acceptable bounds. The use of smaller CMG units should also help alleviate the attitude errors caused by CMG vibration. The solar wings are another source of induced vibration, especially when the LST is traversing from dark to sunlight portions of the orbit. The conventional fix for solar wing vibrations is to make the solar wings stiff so that there is an order of mangitude difference between the control and wing frequencies. Since the LST foldout arrays are now replaced by rollout arrays, the wing frequencies are reduced and could become equal to or less than the control frequency. In such a case, phase stabilization [8] of solar wing vibration becomes feasible. To do this make the control frequency higher than the wing frequency to control the wing motion. During the Phase B activities, additional trades should be made relating the various vibrational modes and sources to pointing performance and selection of the ACS control parameters.

The LST contractor guidelines contain design goals (not firm requirements), such as comet and planet viewing [9]; spectrograph slit orientation; and target positioning in the data field that could impose additional constraints and requirements on the ACS, especially if the ACS were designed to satisfy all functions of the IMC (Image Motion Compensation) system. One major

potential for cost savings is to eliminate the IMC. For this case, additional interfaces between the ACS and experiment sensors would be required. Without IMC, the SSM ACS must provide 0.1 arc second absolute pointing and 0.005 arc second RMS stability. There is considerable question if large SGCMGs can provide this fine pointing because of nonlinearities and resolver inaccuracies. Some recent studies [10] indicate that the SGCMGs must be augmented by a small RW system to obtain the fine pointing performance. Assuming small RWs, then their configuration, size, and use should be the subject of follow-on Phase B studies. For example, the RWs could be used for direct control and the CMGs as a momentum dump (wheel speed control) or the RWs could be used in parallel with the CMGs to provide torque only within the CMG nonlinear zones. The latter case would result in a much smaller RW system. Another arrangement would be to mount one RW on each CMG to compensate for the CMG nonlinearities on a one-to-one basis. Additional considerations that will impact the ACS are expected from the Phase A OTA/SI studies, especially the design and implementation of the OTA FGS and the IMC system.

The MTS is the same as proposed in the Phase A reference. However, the curved EMs could be replaced by bent EMs of similar characteristics if this would simplify manufacturing, as proposed by Ithaco, Inc. Also, additional use could be made of the MTS to damp tipoff rates, to rate stabilize after a misdock, or to provide coarse rate information [11]. To cope with emergencies, these additional functions could be hard wired in the TA [12]. Although MTS have been flown on several small spacecraft, such as OAO, Nimbus, and SAS, none have been built and flown as large as the EM proposed for LST. Early during the Phase B SSM studies, LST size EM should be built and tested for operating characteristics such as power usage, nonlinearities, and field propagation [13].

The RCS selected for the updated LST is similar to the Phase A design reference [2, 3], but a larger tank was selected to permit impulse growth and several redundant items, such as backup thrusters, a pressure regulator, and valving, were deleted. During the Phase B LST study, considerable attention should be devoted to further determining the exact requirements for an RCS. If the rationale for an RCS on the LST should continue to exist, the impulse budget should be kept low enough so that a simple GN_2 system can meet the requirements. From the standpoint of reliability, lifetime, cost, contamination, and toxicity, a GN_2 RCS should be retained for the LST.

CHAPTER XI REFERENCES

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CHAPTER XII. SUBSYSTEMS RELIABILITY

The reliability estimates described in Reference 1 were updated based on the SSM subsystems selected for the updated LST and the new redundancy guidelines. Only that equipment essential to the mission objectives or survival of the LST equipment was considered. Some of the components were unassessed because of insufficient data or indeterminate failure rates. As a result, the reliability numbers for the SSM subsystems will be somewhat lower when these components are taken into account.

Figure XII-1 shows the reliability diagrams used for the control moment gyros (CMGs) and the Digital Processor Assembly (DPA), which are some of the components of the Attitude Control System (ACS). The number shown with each component block represents its corresponding failure rate. Standby failure rates are usually one-tenth of active failure rates. The DPA is an exception to this for which a one-fourth ratio is used. The CMG reliability diagram was one of those proposed by Bendix in their study on LST pointing and control [2]. Figure XII-1 shows that switch reliabilities were included in the CMG computations. The code associated with the numbers that appear in the lower right-hand corner of each block is n/(1+m), where n is the number of elements required, 1 is the number of on-line elements and m is the number of standby elements.

The total ACS reliability diagram is shown in Figure XII-2. In addition to the CMGs and DPA, the ACS includes the rate gyro assembly (RGA), fixed star trackers (FSTs), transfer assemblies (TAs) magnetic torquers (MTs), and the magnetic torquer electronics (MTE). The total ACS has a reliability of 0.98 at the end of 1 year and 0.886 at the end of 2 1/2 years, which is the planned first maintenance visit for return to the ground. It can be seen in Figure XII-2 that the DPA has the lowest reliability number of all components listed.

Since failure rate data for the solar array and battery/charger components presently are not well defined, these were not assessed. A significant amount of redundancy was removed from the remaining components of the Electrical Power System (EPS), reducing the reliability of this system considerably and also having an effect on the overall subsystem reliability. As an example, the number of regulators was reduced from six to four, causing a drop in reliability. It was concluded that the number of regulators required to do the job could be cut from three to two. However, a 2/4 system is still

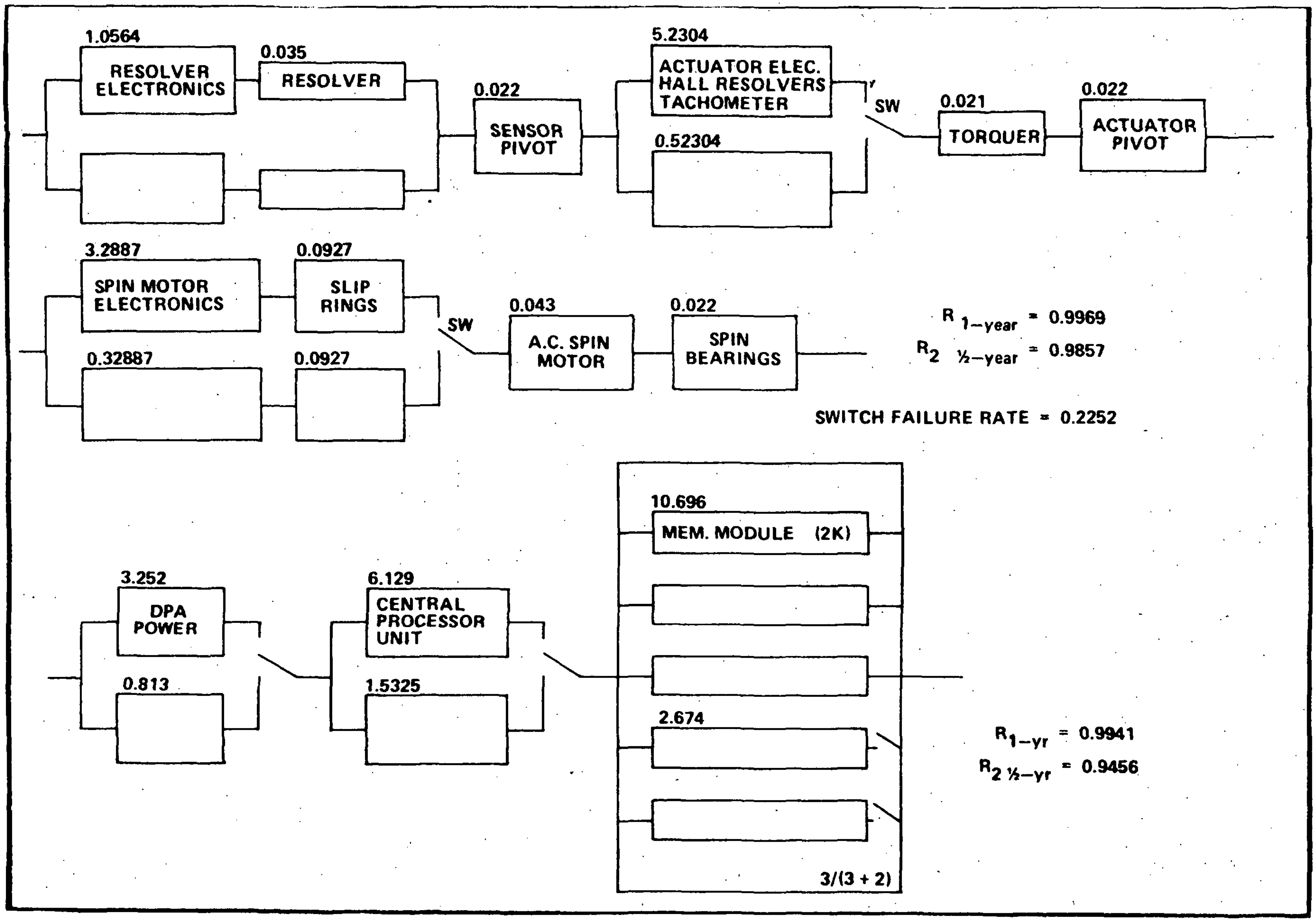
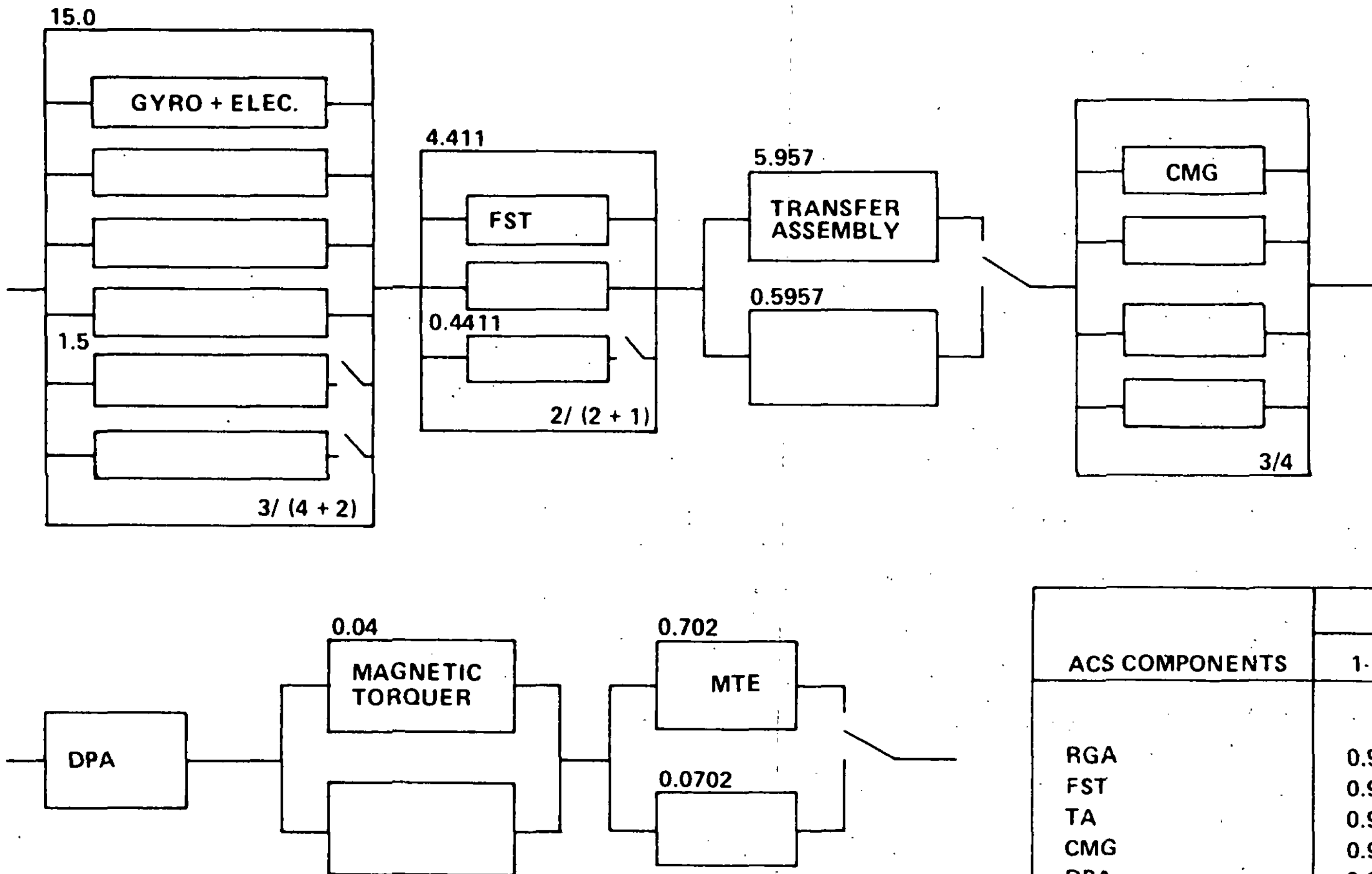


Figure XII-1. CMG and digital processor diagrams.



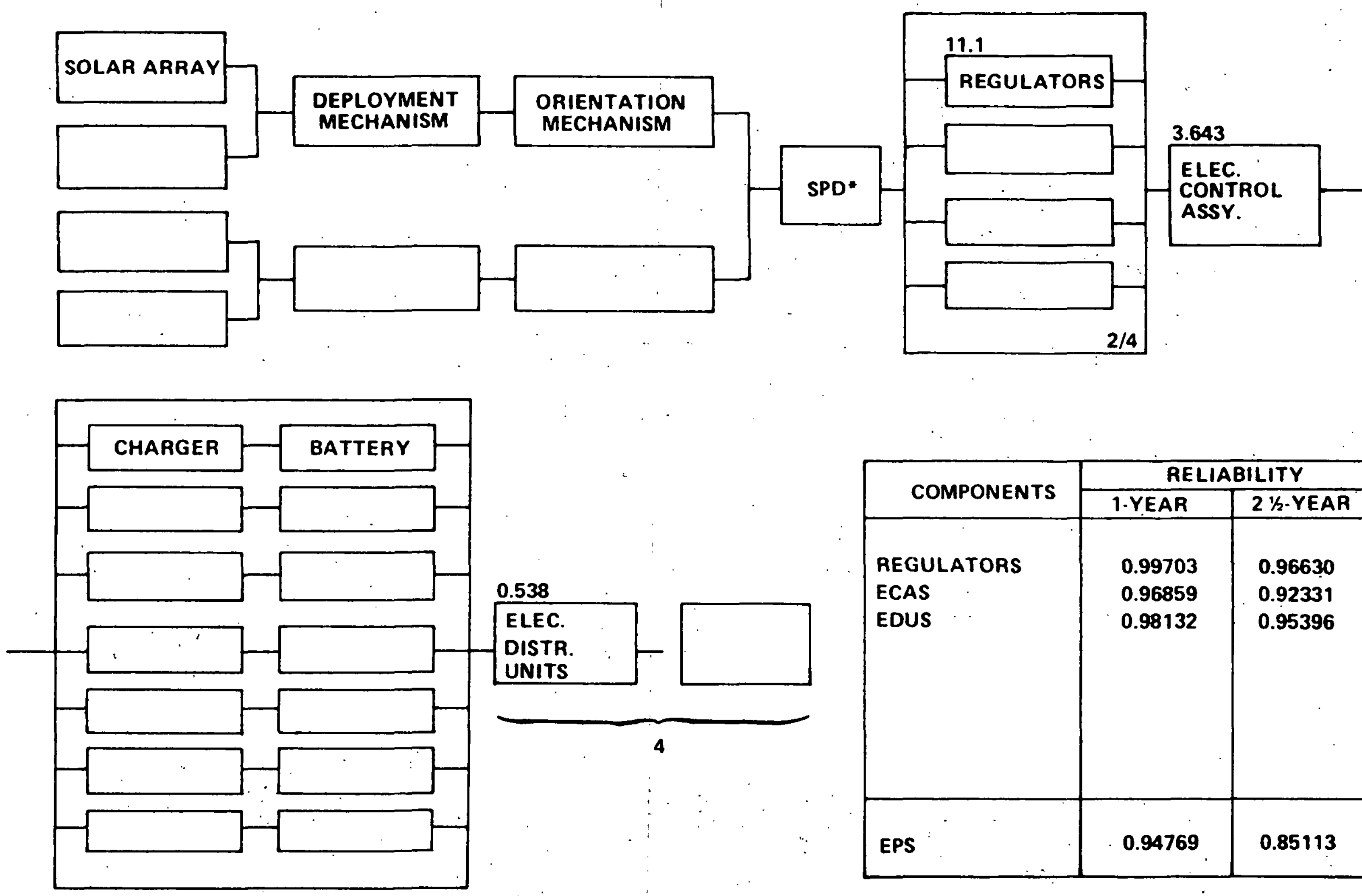
ACS COMPONENTS	RELIABILITY 1-YEAR	RELIABILITY 2-YEAR
RGA	0.99827	0.96240
FST	0.99702	0.98280
TA	0.99855	0.99145
CMG	0.99994	0.99879
DPA	0.99405	0.94560
MT	0.99999	0.99999
MTE	0.99997	0.99987
ACS	0.98783	0.88555

Figure XII-2. ACS reliability.

inferior to a 3/6 system and the reliability number is less than that in Reference 1. Likewise, the Electrical Control Assembly (ECA) and the Electrical Distribution Units (EDUs) were reduced from dual to simplex arrangements. Thus, the overall EPS reliability is 0.95 at the end of 1 year and 0.85 at the end of 2 1/2 years as shown in Figure XII-3.

Arrangement of the Communications and Data Handling System (C&DHS) is such that all of the redundancy shown in Reference 1 is required. With most of the components having some redundancy, this system is highly reliable at the end of 1 year. As shown in Figure XII-4, the C&DHS 1- and 2 1/2-year reliabilities are 0.998 and 0.987, respectively.

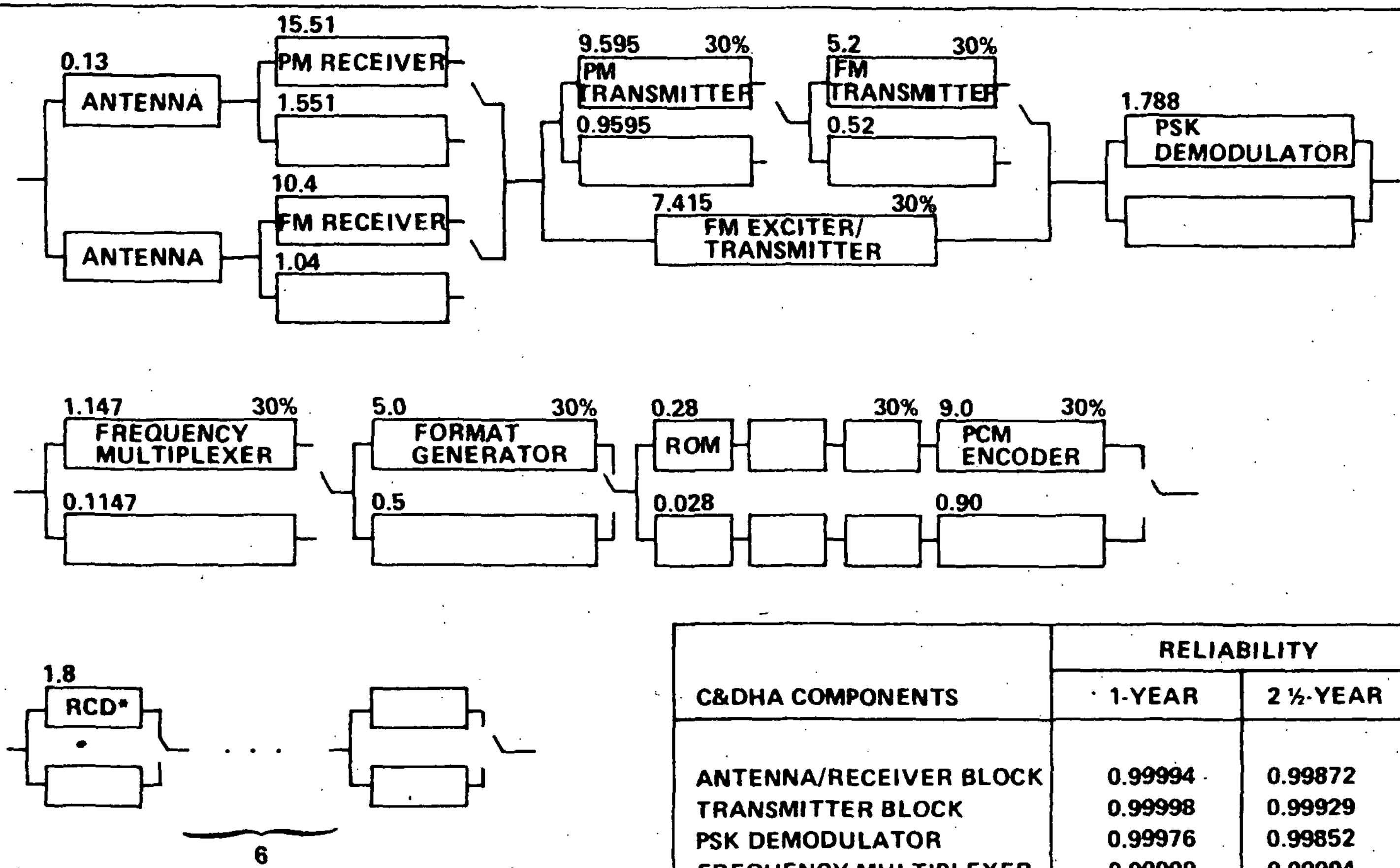
Table XII-1 gives a complete summary of the subsystems reliability at the end of 1 year and 2 1/2 years with reliabilities of 0.93 and 0.74, respectively. The corresponding numbers in Reference 1 are higher with 0.98 for 1 year and 0.91 for 2 years. As stated earlier, the EPS with its decreased redundancy is responsible for the decreased reliability with a drop from 0.998 to 0.947 for the 1-year period. The EPS reliability numbers are believed to be pessimistic and with further in-depth study lower failure rates would result. For reliability calculations the EPS was reduced to a simplex flow diagram, but dual paths exist from source to load in all instances. Although reliability numbers for the EPS are reduced, it is felt that the system is adequate and a higher reliability will exist following additional study. The overall SSM reliabilities are assumed adequate because of the probable on-orbit maintenance visits before refurbishment at 2 1/2 years.



COMPONENTS	RELIABILITY	
	1-YEAR	2 ½-YEAR
REGULATORS	0.99703	0.96630
ECAS	0.96859	0.92331
EDUS	0.98132	0.95396
EPS	0.94769	0.85113

*SOLAR POWER DISTRIBUTION

Figure XII-3. EPS reliability.



C&DHA COMPONENTS	RELIABILITY	
	1-YEAR	2 ½-YEAR
ANTENNA/RECEIVER BLOCK	0.99994	0.99872
TRANSMITTER BLOCK	0.99998	0.99929
PSK DEMODULATOR	0.99976	0.99852
FREQUENCY MULTIPLEXER	0.99999	0.99994
FORMAT GENERATOR	0.99983	0.99899
PCM ENCODER & ROMS	0.99937	0.99619
REMOTE DECODERS	0.99919	0.99502
C&DHS	0.99807	0.98674

*REMOTE COMMAND DECODER NOT INCLUDED: COMMAND PROCESSOR AND MEMORY, DIGITAL ACQUISITION UNITS, TAPE RECORDERS.

Figure XII-4. C&DHS reliability.

TABLE XII-1. SUBSYSTEMS RELIABILITY

Subsystem	Reliability	
	1 yr	2 1/2 yr
Attitude Control (ACS)	0.98783	0.88555
Electrical Power (EPS)	0.94769	0.85113
Communications and Data Handling (C&DHS)	0.99807	0.98674
SSM	0.93435	0.74328

CHAPTER XII REFERENCES

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CHAPTER XIII. LST CONTAMINATION CONTROL

This chapter includes some of the recent experience on the Skylab program which should be applicable to the LST and lists several resultant recommendations for the LST.

Contamination control must be involved in all aspects of LST activity. It must be permitted to influence concept definition and development from a design and operational point of view since contamination has the potential to nullify LST usefulness. Some of the elements of LST activity which must be strongly influenced by effective contamination control are:

- Design
- Selection of materials
- Environmental Control
- Scheduling of events
- Cleaning
- Verification
- Configuration management
- Personnel indoctrination

Since LST mission success may be quite dependent on contamination control, sufficient management emphasis to assure a rigorous pursuit of LST contamination control objectives is required. Based on our current level of contamination knowledge and ATM/Skylab experience, certain management techniques and concepts seem appropriate to emphasize:

- A contamination control advocate should be established at a high enough level of program management or engineering management (perhaps a Special Assistant to the LST Engineering Manager or Program Manager) to assure the constant attention to LST contamination control required in all design and program decisions. This contamination control surveillance should be maintained throughout the LST program including design, manufacturing, testing, operations, maintenance, and refurbishment.

- An LST contamination control specialist should be provided to monitor all operations and tests during assembly of major components, final LST assembly, and launch operations. Quality Assurance personnel involved with other aspects of LST should receive special intensive training in LST contamination control requirements.

- Configuration management should be initiated early in the program to control design and materials in accordance with LST Acceptable Materials and Design Practice Specifications based on contamination control requirements.

- An intensive indoctrination program emphasizing contamination control should be conducted for all personnel associated with the LST.

Much experience has been accumulated with the ATM/Skylab program. Although this program is not complete, nor has all of the contamination data been documented, this experience should be applied to the LST program through the use of the following guidelines:

1. Contamination problems are real and can seriously compromise an experiment unless they are properly considered in design and operational procedure. This must be done early in the design, otherwise extremely costly changes will have to be made. Such changes on Skylab amounted to several million dollars.

2. In general, three types of contamination must be considered: film deposits, particle deposits, and particles in the field of view of the instrument. Techniques for assessing the amount and consequences of each of these types of contamination have been developed for Skylab and are currently being checked against actual data. Film deposits are especially damaging to ultraviolet optics because of absorption and scattering. Good optical surfaces are difficult to achieve for the extreme ultraviolet and only a few monolayers of an organic contaminant could produce severe degradation. Film and particle depositions also produce adverse effects in the visible spectrum, not so much in the reduction of transmission or reflection, but in an increase in the point-spread function which lowers the resolution capability. This is also true for grazing incidence X-ray optical as well as for other imaging systems, such as diffraction limited optics, solar coronographs, and high resolution cameras. Particles floating around the spacecraft can raise the background light level to prevent the observation of faint phenomena. Large (approximately 100 μ) sporadic particles such as have been seen around Skylab are a nuisance to imaging devices, but they can be rejected. However, such particles can

render photometric measurements totally useless unless proper precautions are taken. Star trackers frequently lose their guide star and begin tracking a particle. If extremely small particles (approximately 1 μ) are being continuously produced, a very small amount of total mass can produce a luminous veil around the spacecraft that can prevent any astronomical observations in the sunlit portion of the orbit.

3. Sources of vapor that may deposit on critical surfaces include all nonmetallic materials, particularly paints, RTVs, epoxies, silicones, insulations, plastics, lubricants, etc. Material criteria have been developed to categorize materials into acceptable and unacceptable for space or vacuum use. It should be remembered that such criteria are somewhat arbitrary and should serve only as a guide. Acceptable materials can cause contamination in certain circumstances, and unacceptable materials can be used in certain applications if certain precautions are taken. The biggest danger lies in the indiscriminant use of acceptable materials around optical surfaces. Even though the outgassing rate is low, it is still finite. If the area is large and/or if the temperature is high, significant deposition can occur. These deposits have low vapor pressures and tend to be very presistent once they do condense.

4. Film deposition will result when a cooler surface is exposed to a source in its direct field of view or to another surface which is exposed to a source. The rate of deposition depends primarily on the solid angle subtended by the source, the source material and temperature, and the surface temperature. The best way to prevent film deposition is to baffle or isolate the surface from any sources. If it is necessary to use nonmetallic materials in the vicinity of critical surfaces, their use should be held to a minimum and they should be located to minimize as much as possible their view factor to the surface in question. Extreme care must be taken in the selection of materials with regard to their outgassing characteristics and they should be thoroughly cured and subjected to vacuum soak at elevated temperatures before assembly. If possible, critical surfaces should be designed to run at slightly higher temperatures than their surroundings. Operating temperatures of nonmetallic surfaces should be kept below the temperature of critical surfaces if possible. High temperature operation of components with nonmetallic surfaces must be avoided.

5. Some contamination can be expected on surfaces that do not have any view factor for material sources. Contamination sensors on Skylab looking directly away from the spacecraft collected contaminants at rates from 0.03 to 0.09 $\mu g/cm^2/day$ at temperatures from +10° to -20° C. Sensors with portions of the spacecraft in their fields of view collected at rates of 0.25 to 0.4 $\mu g/cm^2/day$. The material collected by the sensors looking directly away from

the spacecraft is probably outgassing material scattered back by atmospheric drag. This small amount of return does not pose a problem for short term experiments, but will cause significant long term effects unless protective measures are taken, such as covering the optics when not in use, baffling to reduce the solid angle, or orienting the spacecraft such that the optic is not exposed to the velocity vector. Protective covers should also be provided to prevent contamination during ascent, deployment, and rendezvous and docking. The use of GN_2 RCS thrusters does not appear to present a problem. RCS products from the Service Module were detected on Skylab sensors during rendezvous and docking. Sensors which received direct exposure to the plume collected as much as $13 \mu g/cm^2$. Sensors oriented perpendicular to the plume collected $0.3 \mu g/cm^2$ of contamination, but it evaporated almost immediately. The large amount of contamination collected from the direct impingement evaporated at a rate of $6.15 \mu g/cm^2/hr$ from a surface at $0^{\circ}C$. However, because of the corrosive nature of RCS products, sensitive surfaces should be protected from plume impingement.

6. Particle deposits in the form of dust, lint, and other debris are an ever present problem that cannot be avoided, but only minimized. Manufacturing and storage of critical components should be done in clean facilities, but one should not rely solely on the clean facility for protection. Components will get quite dusty after long exposures even in the best clean room. A surface in a class 100 k clean room will collect approximately 3 $\mu g/cm^2$ of dust in a year's exposure, which represents approximately 1-percent surface coverage, just from dust fall. A man wearing clean room garb emits approximately 1 million particles larger than 0.3 μm per minute. Air impinging on a surface will also cause additional particle deposition. Critical surfaces should be always covered when not in use, even in clean storage. The best protection against dust deposition comes from shielding the component from dust fall and by eliminating or reducing the air flow in its vicinity. The most effective means for accomplishing this is to place the component in a container that contains a slight overpressure of clean still air. A continuous purge is not desirable from the point of view of dust suppression because the number of dust particles introduced is the product of the number density and the flow rate. Even class 100 air introduces a substantial number of particles if the flow rate is high. For optics that are affected by dust, some provision should be made to clean them just prior to launch.

7. Care must be taken in storage, transporting and handling, and in vacuum testing spacecraft and their components. It is possible to get a film of material depositing from volatile material used in air-conditioning filters.

Some plastics used from storage have volatile components. Oil films can result from vacuum chambers. Solvents used for cleaning will frequently leave oil films. Normally, the lighter, more volatile deposits do not cause problems because they evaporate rapidly when they are exposed to space vacuum. However, care must be taken when the surface is designed to run cold, or when the surface is exposed to ultraviolet or to a radiation environment. High energy photons (below 3000 Å) or charged particles provide sufficient energy to break molecular bonds and produce chemisorption of the contaminant on the surface. Even highly volatile gases can become permanently adsorbed by this process.

8. Extreme care must be taken with devices such as IR detectors that operate at cryogenic temperatures. The stay time of a water molecule on a surface at LN_2 temperature is longer than the age of the universe. Therefore, ice will accumulate if any water vapor is present in the vicinity of such a device. This has been a problem in the past on Nimbus, although it is believed that the water came from outgassing of the superinsulation near the detector. However, from a large spacecraft, such as Skylab, there may be sufficient backscatter of H_2O molecules in the spacecraft atmosphere to interfere with the use of cryogenic surfaces. Some provision should be made to warm the cryogenic surface to evaporate the ice if it should accumulate.

9. The artificial atmosphere from venting material overboard in the gas phase or from surface outgassing appears to have no effect on any of the instruments on Skylab. Since the molecular speeds are quite high and since the gas expands as an almost collisionless gas shortly after it leaves the spacecraft, the column densities will be extremely low. A detectable effect such as spectral absorption or emission will be very transient and is not likely to cause difficulty. Some homogeneous nucleation may occur in the nozzle, producing particles with dimensions of fractions of microns, but these will dissipate rapidly. A slow leak or material outgassing would be expected to have such a low source rate that detection by scattering, or from adsorption or emission spectra would be very difficult, if not impossible. Therefore, this should not be a source of interference except for the return flux mentioned previously.

10. Control of particles generated in orbit is very difficult. Despite precautions, some particles seem to occasionally come off the spacecraft. These consist of dust and lint trapped on the surfaces or in crevices that slowly work their way out, paint flakes, meteoroid impact debris, ice crystals from overboard dumps, material disintegration, and material and lubricant abrasion around moving parts. Some provision should be made to shield optical surfaces from moving parts, such as protective window opening mechanisms.

11. A number of contamination monitoring instruments have been developed to measure vapor deposition, changes in optical properties, and the size distribution of particulates. Such instruments can be operated under ambient conditions as well as in space and should be designed into the system to provide a continuous history of the state of cleanliness. Having such monitoring can indicate when cleaning is necessary, how the contamination occurred, and how effective the contamination design was. This is as necessary to good contamination control as temperature measurements are to thermal control.

12. Much experience has been gained with contamination control in Skylab and ATM. It appears that the measures taken were adequate and for the most part necessary. Modeling techniques, instrumentation, and control methods have been developed which will apply to all other vehicles. It is imperative that this experience be incorporated into the design of Shuttle and Shuttle payloads at an early state to avoid overly restrictive design decisions as well as design deficiencies.

CHAPTER XIV. CONCLUSION AND RECOMMENDATIONS

Conclusion and recommendations are provided in most of the chapters of the report. The key ones are listed below as an overall summary.

A. Key Conclusions

1. Configuration/Maintenance.

a. Ground maintenance is not a strong driver on the configuration of the LST. Reasonable access is probably the most significant consideration, but this can be sacrificed somewhat, since the LST can be disassembled or access hatches can be provided in the sidewalls.

b. A fairly great degree of on-orbit EVA maintenance is possible, with minimum configuration impact. They key design consideration from the standpoint of EVA maintenance are:

- Access to the equipment, including sufficient spacing between equipment for operation by a gloved hand or special tool.

- Design of electrical connectors and mechanical fasteners to permit operation by a gloved hand or special tool.

- Provision of handholds or footholds to absorb torques and loads generated by replacing the items.

In addition, lighting must be provided for the maintenance operation.

c. Some type of primary frame assembly with integral Shuttle-mounting struts is desirable as a standardized building block for all Shuttle payloads, and is feasible for use on the LST. Although the detailed design of the LST frame may change, the general concept of such a frame should be maintained so that if the evolution of a standardized frame comes about, it can be incorporated easily.

d. Omission of docking for EVA maintenance saves 198 kg (437 lb) of weight on the LST and saves the 1361-kg (3000-lb) weight and 2.2-meter (7.5-ft) length of the docking module in the payload bay. The cost of the docking hardware on the LST and any operational cost associated with the

docking module is also eliminated. The elimination of docking causes no increase in retrieval/holding risk over the ground maintenance approach.

e. One tray concept for either EVA or manipulator maintenance appears feasible. The tray or some similar concept is necessary for manipulator maintenance and becomes necessary at higher levels of EVA maintenance.

2. Structure/Thermal.

a. The open truss SSM structure is not as desirable as the shell concept, except possibly for high degrees of manipulator maintenance.

b. The thermal control system is not impacted significantly by the updated design.

3. Electrical.

a. The rollup array extension/retraction mechanism is more reliable, and the array is more cost effective than the Phase A design.

b. The HEAO batteries can be utilized for the LST.

4. Communications and Data Handling.

a. Circulator switches (nonmechanical) are more reliable than the Phase A design RF switches.

b. The Apollo/ERTS (modified) transponder system should be replaced with an all-ERTS (modified) system.

5. Attitude Control System.

The CMGs can be resized to 203 N-m/s (150 ft-lb/s) from 678 N-m/s (500 ft-lb/s) of Phase A.

6. All Systems.

a. Some redundancy can be deleted from each system in the Phase A design.

b. A high degree of HEAO commonality is still possible in the updated LST design.

B. Key Recommendations

1. Selective use of heat pipes should be studied further for potential cost savings in thermal-vacuum testing and retesting after design changes.

2. Combinations of CMGs and small reaction wheels in several different modes of operation should be studied.

3. Trade studies of the OTA image motion compensation approach versus body pointing of the LST should be pursued.

4. Contamination control experience and recommendations from Skylab and other programs should be utilized.

5. Frequent coordination of the LST Phase B study with the HEAO project and with PD studies on standardized spacecraft and various maintenance modes should be maintained.

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APPROVAL

LST PHASE A DESIGN UPDATE STUDY

By Program Development

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

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